AFIT/GA/GOR/ENY/99M-01

DESIGN AND ANALYSIS OF ON-ORBIT SERVICING ARCHITECTURES FOR THE GLOBAL POSITIONING SYSTEM CONSTELLATION

THESIS

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THESIS

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Degree of Master of Science in Operations Research

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Gregg A. Leisman and Adam D. Wallen

Table of Contents

List of Figures

List of Tables

Table | Page

DEFINITIONS

ALTERNATIVE GENERATION OUTPUTS: Results from defining an alternative servicing system (i.e., number of planes in constellation, servicer capabilities, etc)
CRITICAL COMPONENT: Component whose loss results in mission failure for that satellite
DEPOT: A stockpile of ORU's for any logistical purpose. Can be terrestrial or on-orbit
DEXTEROUS ARMS: The robotic arm(s) that manipulate the GPS satellite in the servicing of selected components.
END EFFECTOR: The "hands & tools" of the dexterous arms that will enable the RMS to open access doors, manipulate thermal blankets, disconnect electrical connectors, unbolt ORU's and handle ORU's. Mostly likely one (or both) dexterous arm(s) will have to be able to use multiple end effectors for the different tasks.
EVALUATION CONSIDERATION: Individual components of a value hierarchy. Any matter significant enough to warrant consideration when evaluating alternatives.
EVALUATION MEASURE: A scale that assesses the degree to which alternatives achieve an objective for a particular evaluation consideration. Also called measure of effectiveness, attribute, performance measure or metric.
GPS SIMULATION INPUTS: Inputs to the simulation that are independent of the alternatives and will likely come directly from interaction with GPS and Aerospace
GRAPPLE ARM: A manipulator that attaches to the GPS S/V and repositions the dexterous arms to the work site. For University of Maryland's Ranger Program this is a 7-Degree of Freedom manipulator.
LAYER or TIER: A set of evaluation considerations a uniform distance from the top of a value hierarchy.
LOGISTICAL & TRANSPORTATION SYSTEM (LTS) - This is the entire Robotic Servicing System (RSS) minus the Robotic Servicer. This includes the launch vehicles, the ORU transport system, and the different orbits to get the ORU's from the ground to hand-off with the RS.
MEASURE WEIGHT: A quantification of the relative impact of a particular measure on the overall value model.
OBJECTIVE: The preferred direction of movement associated with an evaluation consideration.
ORBITAL REPLACEMENT UNIT (ORU): A component or black box on the GPS satellite vehicle (S/V) that will be removed and replaced. Since most GPS components are packaged in electrical boxes, they can also be called black boxes; however, they could also represent non-box like components like reaction wheels.
ORU (Orbital Replacement Unit): The component that will be added or exchange on the user S/V for the purpose of maintenance, upgrade or retrofit. ORU's can come from terrestrial venders or dead user S/V's.
OTC (ORU Transport Container): The physical container to transport ORU's from earth to orbit for rendezvous with the RSS. Can be piggybacked on other launched or the payload for a dedicated launch.

PIGGYBACK: Attaching a small OTC to another payload to be lifted on-orbit. While preferably the payload would be a GPS S/V it could be any acceptable payload.
PAYLOAD: 'Payload' refers to the mass a launch vehicle delivers to orbit. 'Final payload' refers to mass delivered by the LTS into the target orbit.
PARKING ORBIT: The orbit into which the launch vehilce would place the canisters or robotic servicers. A parking orbit will be used if the canisters or robotic servicers have their own upper stages to insert themselves into the target orbit.
POSITIONING ARM: A robotic arm that moves the RS to the work-site once the RS and GPS S/V are docked.
ROBOTIC MANIPULATING SYSTEM (RMS): The RS's payload, which includes the dexterous arms, end efforts, robotic vision system (RVS), grapple arm or positioning arm (if needed), and the task interactive computer (TIC). Its function will be to service the user S/V and also possibly perform the docking functions.
ROBOTIC SERVICER (RS) – The entire spacecraft that does servicing, including the RMS, the docking unit, the bus, and propulsion unit. The RS is a sub-component of the overall Robotic Servicing System (RSS).
ROBOTIC SERVICING SYSTEM (RSS): The entire servicing infrastructure, including the RS, the canisters, launch vehicles, dispensers, upperstages, and ORU's.
ROBOTIC VISION SYSTEM (RVS): The video camera(s), the camera's support arm(s), lighting, and other sensors needed for the RMS to perform its duties
RS BUS: The subsystems on the RS that provide power, ground communication, navigation, attitude control, close proximity maneuvering, and the ORU Storage System (OSS).
RS TRANSPORT VEHICLE (RTV): For the free-flying servicer configuration, this would be the "mother" vehicle for the SMS. This provides the power generation, data management, orbital maneuvering propulsion system, ORU pallet, and the bulk of the data management and communication.
ROBOTIC SERVICING SYSTEM (RSS): The entire infrastructure to install ORU's on the user S/V. Does not include servicing modifications to user S/V.
S/V: Satellite vehicle
SCORE or LEVEL: The rating for a particular alternative with respect to an evaluation consideration.
SERVICING MICRO-SATELLITE (SMS): Composed of the RMS and support systems, this satellite would detach from the RS and service the GPS S/V. The benefit of this configuration is this could be much smaller than the entire RS and thus more easy for docking and servicing. (Not used in every alternative)
SINGLE-STRING FAILURE: Last redundancy level for a component fails
SUB-VALUES: Focused elements of values and other sub-values. Everything between the values and the measures.
TARGET ORBIT: The destination orbit for the LTS system. This will be the operational orbit for the robotic servicers, and the orbit by which the ORU's will be picked up for the canisters.
TASK INTERACTIVE COMPUTER (TIC): The processors that control the manipulators. Whether automated or teleoperated, operational robots have a feedback loop that is not feedback to the user. The reciprocal of the TIC is the ground-based Human-Interactive Computer (HIC). The HIC sends the appropriate commands from the human operators to

the RS.
VALUE HIERARCHY: A value structure organized in a hierarchical manner.
VALUE MODEL: The value hierarchy combined with the value functions and measure weights.
VALUE STRUCTURE: The structure that reflects the decision-maker's objectives and priorities.

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Abstract

Satellites are the only major Air Force systems with no maintenance, routine repair, or upgrade capability. The result is expensive satellites and a heavy reliance on access to space. At the same time, satellite design is maturing and reducing the cost to produce satellites with longer design lives. This works against the ability to keep the technology on satellites current without frequent replacement of those satellites. The Global Positioning System Joint Program Office realizes that it must change its mode of operations to quickly meet new requirements while minimizing cost.

The possibility of using robotic servicing architectures to solve these problems is considered in this thesis. The authors accomplished this through a systems engineering and decision analysis approach in which a number of different alternatives for on-orbit satellite repair and upgrade were analyzed. This approach involved defining the problem framework and desired user benefits, then developing different system architectures and determining their performance with regard to the specified benefits. Finally, the authors used decision analysis to evaluate the alternative architectures in the context of the user's goals. The results indicate favorable benefit-to-cost relationships for on-orbit servicing architectures as compared to the current mode of operation.

DESIGN AND ANALYSIS OF ON-ORBIT SERVICING ARCHITECTURES FOR THE GLOBAL POSITIONING SYSTEM CONSTELLATION

I INTRODUCTION

1.1 Background

Satellites are the only major Air Force systems with no routine repair, maintenance, or upgrade capability. This has motivated the space community to build extremely reliable, redundant satellites, and replace them the first time one critical component fails. The result is expensive satellites and a heavy reliance on access to space.

The majority of weapon systems take advantage of the capabilities that satellites lack. Aircraft last much longer because of routine maintenance. Aircraft are not discarded at the first subsystem failure; instead, maintainers repair them. Most importantly, aircraft usefulness is extended by payload upgrades; examples include the EF-111, Wild Weasel, JSTARS, F-15 Strike Eagle, and many other weapons systems both in and out of the Air Force. With the ability to provide logistics to the end system, the Air Force is much more efficient at developing, maintaining, and operating its aircraft systems than its space systems.

Is there a feasible, cost effective approach to applying logistics to space systems? Certain technologies and infrastructures are maturing enough to make some in the satellite community answer this question in the affirmative. First, the computer and information revolution has made automation and teleoperation feasible for space operations. Systems are already under development for the international space station. NASA, the University of Maryland, and other

universities will be demonstrating telerobotic servicing in the Space Shuttle payload bay in the year 2000 with the Ranger Telerobotic Shuttle Experiment (RTSX). Second, new space propulsion technologies may offer cost benefits for various types of missions. Space technologists have explored Reusable Orbital Transfer Vehicles (ROTVs) for years, and new technologies such as electric and solar thermal propulsion will dramatically change the propellant budgets for orbital transfer. Finally, space programs and their supporting infrastructures have matured enough to provide the necessary economies of scale to make satellite servicing cost effective. Factors such as large satellite constellations in specific orbits, mass manufacture of satellites, and routine launches to certain orbits make on-orbit servicing more feasible.

1.2 Problem Statement

Satellite programs have to wait extended periods of time to implement significant capability upgrades. Such upgrades must wait for their incorporation on new replacement satellites. A satellite program will typically only launch new satellites when the existing satellites fail at the end of their design life. The wait this requires is getting progressively longer as satellites achieve longer and longer design lives. Without repair, it is expensive to maintain a satellite constellation through a policy of launching a new satellite at the first on-orbit failure. Increasing satellite design life is a common strategy to combat this problem. However, this strategy leaves the satellite program in a perpetual state of being technologically out of date. In the battlefield and marketplace of the near future, this can mean employing outdated and expensive systems against satellite programs that do a better job of rapidly and efficiently implementing the latest advancements. The Air Force needs a fast, flexible, and cost effective system of upgrading and maintaining a constellation of satellites.

1.3 Scope

This research used the Global Positioning System (GPS) constellation as its primary case study. However, we tried to always approach the problems and their potential solutions from the perspective that the results should be broadly applicable. As shown in Chapter 2, trying to develop accurate characteristics of a servicing system for any and every operational satellite produces an unreasonable level of complexity. By using one case study, we were able to generate and evaluate realistic solutions. Due to its size and orbital requirements, an on-orbit servicing system for GPS would be applicable to many different satellite systems. Therefore, another satellite program could modify our analysis to their needs with little difficulty. Our objective in this research was to demonstrate whether on-orbit servicing will be a viable improvement over the current way of doing business.

We explored the current constellation design as a baseline alternative. The remaining alternatives consisted of the current constellation with the addition of various servicing architectures. We assumed no radical shift in GPS management policy. For example, we did not consider eliminating the GPS constellation by adding its payload to another platform such as the Space Based Infrared System. In addition, alternative generation did not consider changes to many of the parameters of the current constellation such as the number of satellites or the current 12-hour orbital period. Our sponsor gave us latitude to analyze a three-plane constellation, since this does not effect GPS's coverage significantly. We proposed only technologies that are currently in operation or development. Their application may have deviated from the current intent, but all the technologies presented were already under development when this research began.

1.4 Assumptions

The assumptions below cover the general aspects of this research. Where applicable throughout this paper, we have included more specific assumptions.

1. The technologies we discussed in this paper will continue to develop and receive funding.
2. The Space Shuttle has an altitude range of a few hundred kilometers; however, GPS's orbital altitude is over 20,000 kilometers. Therefore, we assumed manned servicing would be impractical. In addition, due to the results of earlier studies, we did not consider retrieving GPS satellites to Low Earth Orbit (LEO) (see Ch. 2).
3. We only utilized current or approved launch vehicle programs (see Section 4.4.3 for further details).
4. Satellite program directors see the need for a change to the status quo regarding how they manage their satellite or constellation of satellites.

1.5 Document Overview

Chapter Two presents a review of past relevant literature about on-orbit servicing. It brings the reader up to date on recent developments in the satellite community and includes information about pertinent Air Force initiatives. Chapter Three provides an explanation of the methodology that was the foundation of this thesis effort. Chapter Four contains results and their accompanying explanation and analysis. Finally, Chapter Five presents the authors' conclusions.

II Literature Review

2.1 History

2.1.1 Introduction

The historical research in on-orbit servicing falls into two categories. The more obvious category is the research into what can be done. By pushing the use of newer technologies, mankind's benefits from space systems has only multiplied. However, a second equally important category is the systems analysis of what should be done. There is an inherent interplay between the two categories. As new technologies become available, organizations question whether there are more beneficial ways of achieving their goals. However, as analysis illuminates benefits to new systems, people will push for research into new technologies. On-orbit servicing has experienced the same push and pull history.

On-orbit servicing has been in existence for over twenty-five years. In 1973, astronauts demonstrated the feasibility of on-orbit servicing when they serviced the Skylab Space Station on orbit. These repair missions included release of one of Skylab's solar arrays, deployment of a makeshift sun shield, and repair of critical bus components including a microwave antenna and rate gyro package (Waltz, 1993, 10). Another example was the 1984 rendezvous, capture, repair and redeployment of Solar Max by the Space Shuttle.

2.1.2 Space Assembly, Maintenance, and Servicing

A fair portion of the systems analysis of on-orbit servicing was done in the mid to late 1980's. With Skylab, Solar Max, and other servicing missions establishing the feasibility of on-orbit servicing, U.S. space agencies began to question if the benefits of on-orbit servicing could be incorporated into the majority of space missions. The need for an answer was magnified by the U.S.'s position to develop new large space systems like Space Station Freedom, and the

space based Strategic Defense Initiative (SDI). These programs would be extremely expensive without an efficient way of providing space logistics. Therefore, in 1986, the Department of Defense and the National Aeronautics and Space Administration (NASA) initiated a joint study called Space Assembly, Maintenance, and Servicing (SAMS). "Its primary objective is to define and establish, where cost-effective, SAMS capabilities to meet requirements for improving space systems capability" (Waltz, 1993: 5). TRW and Lockheed performed Phase I of the SAMS systems analysis study from February 1986 through June 1987. SAMS identified that astronauts or robots could perform on-orbit servicing. However, due to the technological limitations in the mid 1980's, SAMS outlined both a manned servicing system and future growth capability of space robotics (Waltz, 1993:42). With a near-term objective of servicing a myriad of large space systems, SAMS identified the need for a manned servicing architecture. This architecture included the following: servicing facilities at Space Station Freedom, a reusable orbital transfer vehicle (ROTV) that would use cryogenic propellants, an on-orbit storage facility for ROTV's propellants, a smaller orbital maneuvering vehicle (OMV), and a variety of other capable and robust space systems (Waltz, 1993: 253). Not surprisingly, this was not a low cost architecture. The proposed architecture would amortize its large expense over multiple satellite programs. SAMS found nominal life cycle cost savings of around 20 to 30 percent. A limited number of programs had cost savings of up to 50 percent, and others had no savings at all (Waltz, 1993: 238, 245). With its huge infrastructure costs, the SAMS architecture was never adopted; however, SAMS provided a good baseline study from which other conceptual research could proceed. Some researchers attempted to place numbers on the performance and costs of servicing and concluded that servicing's time had not come (Forbes, 1988: 10). One researcher focused on the potential benefits of refueling (Hotard, 1989), and another stressed the importance

of designing satellites for on-orbit servicing (Wyatt, 1987). Each of these efforts offered models of how to evaluate servicing as a satellite management alternative. For some researchers, the main focus of their work was developing a methodology to perform such an evaluation. (Del Pinto, 1988)

2.1.3 Orbital Maneuvering Vehicle – Flight Telerobotic Servicer

One of the initiatives that coincided with the SAMS study was the Orbital Maneuvering Vehicle (OMV) Flight Telerobotic Servicer (FTS) system. FTS began in 1986 as a capable, human-like, telerobotic servicer for the space station (Waltz, 1993: 202). The OMV was an orbital transport system that could operate from the space station, space shuttle or an aerospace plane and serve a wide variety of customers (Wyatt, 1987: 11). Attached to an OMV, the FTS could service low earth orbit (LEO) satellites. Unfortunately, where robotic technology was immature, OMV-FTS compensated with size and cost. Thus, with the cancellation of SDI, and the realignment of the space station objectives, Congress cancelled funding for FTS in 1991 (Spencer, 1991).

2.1.4 Basic Research

As the nation shifted priorities away from large space systems in the early 1990's, large expensive robotic servicing systems lost their support. Thus, during the 1990's, researchers have focused on how to perform robotic servicing better and cheaper. The research quickly identified teleoperation as a key issue in any kind of unmanned satellite servicing domain. As part of its telerobot testbed, the Jet Propulsion Laboratory (JPL) developed software to simulate interaction between a servicer and a satellite. The simulation helped human planners understand the difficulties of such interfacing (Mittman, 1988, abstract). JPL is NASA's lead center in telerobotics and continues to investigate teleoperation and remote system automation. It has

performed research in manipulator modeling and control, real-time planning and monitoring, real-time sensing and perception, and overall system architectures. JPL has also surveyed the possible applications for space robotics (Weisbin, 1991, abstract). In both size and cost, this basic research has paid off. First, program cost has dropped from the hundreds of millions for the FTS, to the tens of millions for a current servicer called Ranger (see Sections 2.2.2 and 4.4.9). Second, size has also dropped dramatically. An excellent example of this is the decrease in size for Mar's rovers from the planned refrigerator size in the 1980's to the current Mars Pathfinder, which is the size of a trash can.

2.1.5 Hubble Space Telescope – The Design of a Serviceable Satellite

During the time that robotic servicing shifted into basic research, NASA has forged ahead with application of manned servicing. NASA engineers designed the Hubble Space Telescope (HST) under a philosophy based on modularity, standardization and accessibility. (Smith, 1986: 7) Early in the design process, the designers identified candidate items for servicing. Candidate selection was based on likelihood of failure during HST's planned lifetime, criticality to telescope operations, accessibility, and cost of servicing. The engineers designed these parts as modular Orbital Replacement Units (ORUs).

> These units include critical subsystems for spacecraft operation and science data collection, as well as candidates for future upgrading. Most of these modules are self-contained boxes that are installed or removed by simple fasteners and connectors. ... 26 different components, some duplicated to make about 70 individual units, were selected as ORUs. These include the telescope's batteries, fine guidance sensors, solar arrays, computers, reaction wheel assemblies, and other major system components, as well as the focal plane detectors and cameras. The ORUs range in size from small fuses weighing only a few ounces to 700-pound scientific instruments as large as a telephone booth. (Smith, 1986: 7-8)

By standardizing common elements such as bolts and connectors, HST's designers reduced the number of unique components and tools needed for servicing missions. For example, the design included 7/16-inch double-height hex to hold all of the ORUs in place. Thus, the astronauts

needed only one tool to remove and install them (Smith, 1986: 8). The designers also recognized the importance of accessibility. Most of the ORUs are in equipment bays around the perimeter of the spacecraft. Large doors open to reveal these bays for ORU inspection and handling of the units (Smith, 1986: 8-9).

2.2 Recent Developments

Events in the last ten years have changed the landscape considerably since the SAMS study performed a systems level analysis of on-orbit servicing ten years ago (section 2.1.2).

2.2.1 Results of Hubble

By implementing the servicing concept, Hubble is able to upgrade its instrument payload every five years. In addition, servicing has given Hubble an unprecedented twenty-year mission life (Waltz, 1993:13). More importantly, on-orbit servicing has transformed Hubble from being a billion dollar failure, because of its flawed mirror, to an extremely successful program that is expanding the knowledge of mankind.

2.2.2 Current Robotic Servicing Initiatives

While Congress cancelled funding for FTS in 1991, the need for on-orbit servicing of the space station did not go away. Canada picked up the slack by approving the development of the Special Purpose Dexterous Manipulator (SPDM) (Asker, 1997: 73). The SPDM will perform servicing on the International Space Station (ISS) in lieu of much of the astronaut extravehicular activity (EVA) work. Another international on-orbit servicing project named JERICO was a joint venture between the European Space Agency (ESA) and Russia. The JERICO project was to install a robotic servicer on the outside of the Mir Space Station (Didot, 1996). Unfortunately, the recent history of Mir has precluded the implementation of JERICO. Japan's National Space Development Agency (NASDA) launched an experimental satellite last year named ETS-VII.

NASDA intended to verify the following technologies: unmanned rendezvous and docking, cooperative control between satellite attitude and robotic arm motion, teleoperation, and other technologies basic to on-orbit servicing (Matsue, 1996: 536). The Ranger Telerobotic Servicer is a focal point for U.S. space robotics. Ranger's engineering and science objectives are to quantify the capabilities of space telerobotics and correlate them to Earth-based simulations ("Ranger Program Overview," 1997). As robotic servicing programs demonstrate their growing capabilities, they justify reassessment of on-orbit servicing as a satellite management alternative.

2.3 Air Force Initiatives

In 1995, then Air Force Chief of Staff General Fogelman tasked the Air University to look thirty years into the future to identify the concepts, capabilities, and technologies necessary for the United States to remain the dominant air and space power in the next century. The resulting study was Air Force 2025. In its overview document, the study identified a migration in air force operations from primarily an air focus to primarily a space focus (2025, Overview, "Trends"). Volume II of Air Force 2025 devoted chapter three to "2025 Aerospace Replenishment: The Insidious Force Multiplier." It mentioned that space replenishment could extend the useful life cycle of satellites. This offered the benefits of reducing launch costs, reducing satellite replenishment costs, and reducing space debris (2025, Vol. 2, Ch. 3, "Introduction").

> The aerospace replenishment mission statement for operations in 2025 is to provide on-demand support to air and space vehicles requiring replenishment. Global aerospace mobility and on-demand support are the aerospace replenishment core competencies required to support air and space operation in 2025. (2025, Vol. 2, Ch. 3, "Core Competencies")

The study identified satellites' primary need as refueling with a secondary benefit of such a capability being the upgrade of satellite components if satellite and component makers transition

to on-orbit accessible modular designs (2025, Vol. 2, Ch. 3, "Space Support System"). Chapter Five, "Spacelift 2025: The Supporting Pillar for Space Superiority," of Volume II advocated routine space operations as the only way to accomplish our future air and space objectives.

> There simply is not enough funding available to develop innovative space-based capabilities while continuing to employ brute force methods of getting to orbit. Routine operations are more affordable, because they eliminate the large standing armies required by the research and development (R&D) processing philosophy of current expendable systems. (2025, Vol. 2, Ch. 5, "Introduction")

The study proposed that Orbital Transfer Vehicles (OTVs) should be an integral part of this new system. Their primary mission would be to receive satellites launched to low earth orbit (LEO) and then move the satellites to their final orbits, but they could also add life to satellites by refueling, rearming, and re-supplying them. OTVs could also protect the US space architecture (2025, Vol. 2, Ch. 5, "Introduction").

The Air Force must refine and add specificity to these general statements regarding future management of our space assets. We must take these requirements and objectives, develop alternatives, and systematically evaluate their benefits and deficiencies. Satellite program managers can use the resulting analysis to make informed decisions about the course of research, development, and implementation in this arena. Several approaches are useful for addressing these complicated issues. We have chosen to use a combination of the systems engineering and decision analysis approaches. Our approach began with a top-level identification of the issues important to the decision-maker. Then, we worked with the decision-maker to develop metrics which enabled us to evaluate the performance of each alternative we developed. When this evaluation was complete, we conducted an in depth analysis of the results and presented the conclusions to the decision maker. The tools we developed through the course of this research could be useful for future analysis of unexplored alternatives and flexible enough to reflect the knowledge and technology of an evolving space logistics environment.

III Methodology

3.1 Overview

3.1.1 Decision Analysis

Everyone encounters decision-making on a daily basis. We make decisions in an environment filled with information that includes current data, historical data, advice from others, and personal perceptions. There is potential for uncertainty in all of this information. For example, we decide what time to set our alarm based on how long it typically takes us to get ready in the morning and how long we expect the trip to work to take. We decide what to have for breakfast based on resource availability. How much time did I allot for preparing and eating breakfast? Do I have a filter and grounds to make coffee? Do I have enough milk for a bowl of cereal? How old is the milk? Should I fix some eggs or do I need them for tonight's dessert? Will I have time to pick up more eggs on the way home? We decide which route to take to work based on the weather, traffic reports, and which way has proven to be quickest under the current circumstances. In the case of an AFIT student or professor, we also choose our route to work based on the time we expect to arrive, the gate schedule, and the assumption that the gate schedule didn't change again. These three decision-making examples occur daily, and we haven't even started the workday, yet.

Decisions such as these daily occurrences give us little or no pause. There are two reasons for this. First, we have made these decisions so many times that we understand the context in which we make them. In other words, we have high confidence that we understand and will accurately interpret the information available to us. Second, we understand the possible outcomes of our decisions. Often times, neither of these conditions exists. Choosing when and where to take a family vacation, how to make personal investments, and whether or not to stay in

your current job are all difficult decisions. Clemen in Making Hard Decisions: An Introduction to Decision Analysis (Clemen, 1996) identifies four basic sources of difficulty. First, a decision can be hard due to its complexity. It may be impossible or impractical to investigate or even identify all the alternatives. There could be significant social or economic impacts resulting from the final decision. Simply identifying the key influential factors may be a daunting task. Decision Analysis (DA) provides tools to deal with the complexity of a decision problem. The process of creating a value model can help the analyst and the decision-maker (often these are not the same individual) identify the key elements of a situation that can affect the decision. The value model assists understanding the relationship between alternatives and their overall value to the decision-maker. Later in this chapter, we will explain these concepts and how we use them.

Decision difficulty may also arise from inherent uncertainty. In many problems, uncertainty is the central issue. The performance of particular alternatives may contain considerable variability, which, in turn, introduces significant variability in the possible outcomes. DA allows us to represent this uncertainty in a systematic and meaningful way. Thus, we are able to draw meaningful conclusions from the results. Clemen's third source of difficulty in decision making is the possibility that there are multiple objectives. Often, objectives compete with each other. Accomplish one objective may necessitate compromising on another. For instance, businesses want to minimize costs while maximizing profits, but a business will incur cost to generate revenue. The challenge, then, is to assess the alternatives and determine which one results in the most agreeable compromise between these two objectives. Another objective in the same problem may be to avoid exceeding a certain level of acceptable risk. This may pull the decision in a completely new direction. It is important to consider all the objectives important to a decision-maker. The DA approach provides a framework and tools to

account for multiple objectives. The possibility that different perspectives lead to different conclusions is the fourth basic source of decision-making difficulty. This is primarily a concern when there are two or more people involved in making the decision. Each individual's perspective may result in a wide range of perceptions concerning the value structure of the problem or the uncertainties of the alternatives and possible outcomes. The way to combat these issues is to involve the decision-making team early in the DA process. They should play an active role in the value structure development and throughout the process whenever possible (Clemen, 1996: 1-2). Figure 3.1-1 comes directly from page 6 of Clemen, and shows the process's underlying methodology.

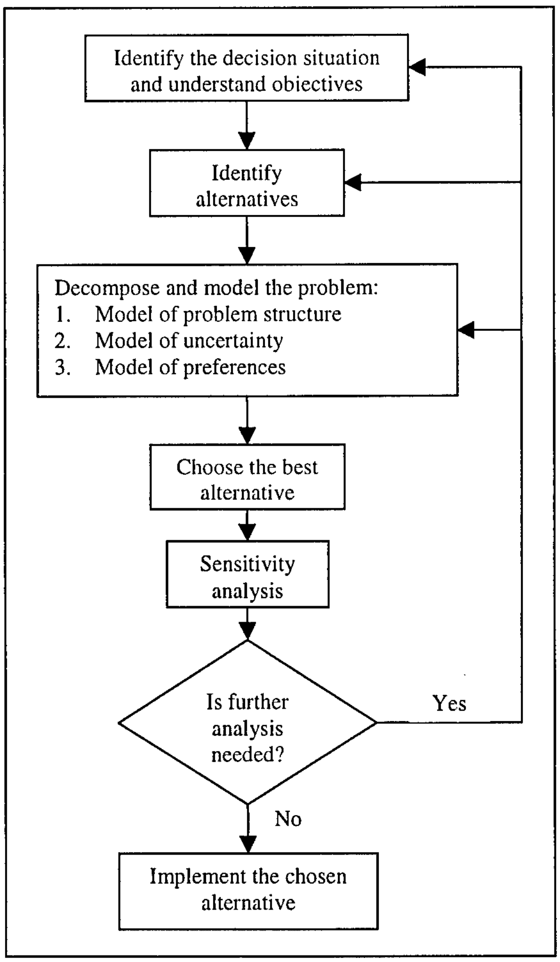

Figure 3.1-1. Decision Analysis Process Flowchart

The first step of the process is to represent the decision situation and objectives through the development of a value model. Alternative identification is next, and detailed modeling of the problem's structure, uncertainty, and preference follows. The model yields a quantification of the alternatives, which the analyst then evaluates. Before presenting the results to the decision-maker for implementation, it may be necessary or beneficial to perform multiple iterations of the process up to this point (Clemen, 1996: 7).

3.1.2 Systems Engineering

Systems Engineering (SE) is similar to decision analysis in that it uses an orderly process to solve a problem. Users of SE apply it to engineering applications, in efforts to manipulate the physical environment to solve a problem. Systems engineering is as concerned with the structure and behavior of the system as with the mathematical techniques to design the solution. Like decision analysts, systems engineers understand that the system complexity requires the engineer to develop a clear detailed process description. Hall's systems engineering description is a good framework to understand the SE process (Hall, 1969). In one dimension, Hall outlines the traditional management of a program. Steps in this dimension include program planning, project planning, development, production, distribution, operations, and retirement.

The second dimension is the real benefit of the SE process. Within each management step there is a logical process to find the best solution for that step. The steps in the logical process are problem definition, value system design, system synthesis, system analysis, rank (or optimize) the alternatives, decision-making, and planning for the future (Hall, 1969). Problem definition requires the systems engineer to determine what is and is not the problem he or she is trying to solve. This step involves defining the needs, alterables, stakeholders, constraints, scope and future environment of the problem. Value system design identifies the system objectives

and evaluation criteria. System synthesis is the generation of different feasible solutions to the problem. System analysis involves modeling the different alternatives to understand their different characteristics. The next step is to evaluate the different alternatives in a way that makes it possible to rank order them. With these evaluation results, the systems engineer will present them to the customer for a decision. The last step is to communicate the results and develop a plan to implement the solution.

With all of this information we still lack one very important piece. What is the key decision we are trying to answer? The question is this: Should the GPS Joint Program Office (JPO) view a satellite management system of on-orbit servicing as an alternative to its current system of phased upgrade through replacement? To answer this question, we combine the processes of decision analysis and systems engineering. In this way, we are able to combine our individual disciplines to answer an extremely complex question. Our combined process is similar to both the DA and SE processes since a fundamentally logical problem solving technique is common to both. The combined process is in the diagram below.

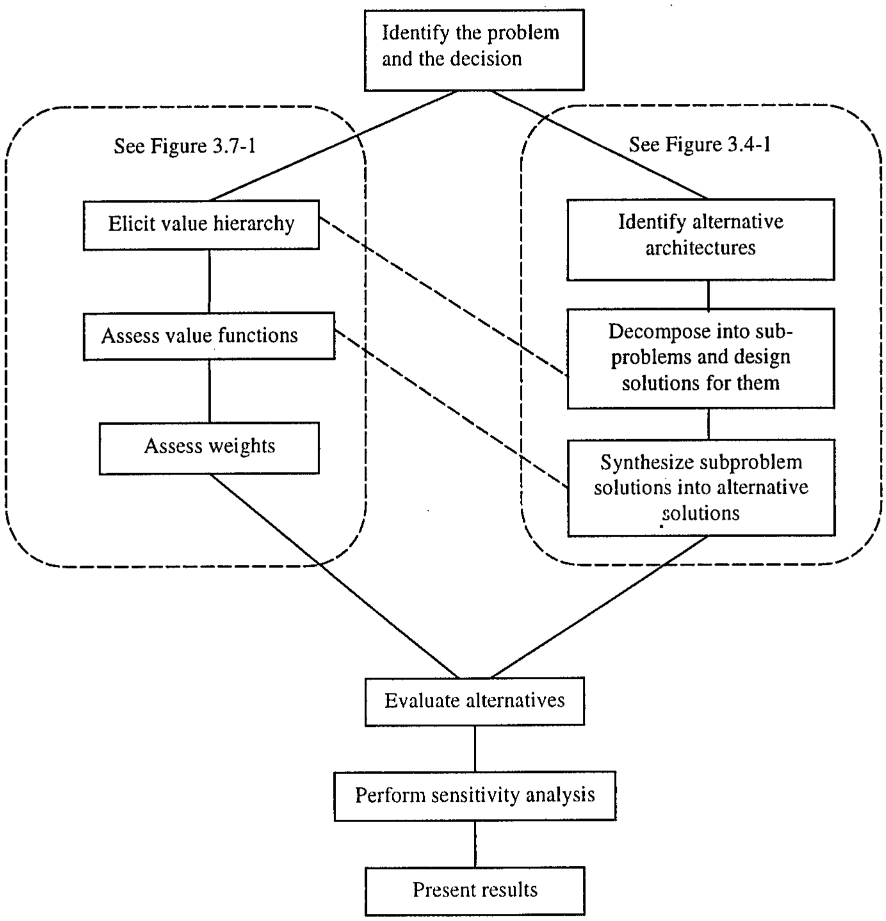

Figure 3.1-2. Composite Process Diagram

3.2 Identify the Problem and the Decision

The first step required that we identify the major problems, sub-problems, and the relationship between them. Next, we characterized the problem by describing the constraints, alterables, stakeholders, and future influential conditions. In addition, we needed to identify the major subjective considerations. A final important problem identification step was to identify the scope of the problem.

3.3 Elicit Value Hierarchy

A critical step of the framework was the elaboration of the value model. The choice of the word "elaboration" instead of "definition" is significant, because the latter implies the

creation of something new, while the former implies putting onto paper something that already exists. Certainly, the decision-maker made decisions before developing a value model. When he made those decisions, he based them on his objectives and priorities for his organization. By explicitly stating these objectives and priorities in a structured manner, it was possible to evaluate alternatives and make recommendations that reflect the decision-maker's pre-existing decision making process.

The structure that reflects the decision-maker's objectives and priorities is called the value structure. When this is organized in a hierarchical manner, it is called a hierarchy. The individual components of a value hierarchy are evaluation considerations. An evaluation consideration is any matter significant enough to warrant consideration when evaluating alternatives. The hierarchy is organized into layers or tiers, where each layer consists of the evaluation considerations a uniform distance from the top of the value hierarchy. Thus, the first-tier evaluation considerations are those in the first layer below the top of the value hierarchy, the second-tier considerations are two layers from the top, and so on. Each evaluation consideration has an associated objective, which is the corresponding preferred direction of movement. For example, flexibility is an evaluation consideration in this value hierarchy, and the objective is to achieve increased flexibility. An evaluation consideration may be associated with a goal. A goal is a threshold of achievement for alternatives. Evaluation measures are scales that assess the degree to which alternatives achieve an objective for a particular evaluation consideration. This is sometimes called measure of effectiveness, attribute, performance measure, or metric. The level or score is the rating for a particular alternative with respect to a specified evaluation measure (Kirkwood, 1996: 11-13). The value model is the hierarchy with the value functions and measure weights. The value functions translate scores for a particular measure into value,

and they come directly from the decision-maker. The weights reflect the significance each measure has when evaluating an alternative. The weights also are elicited directly from the decision-maker. This is the terminology we used to describe GPS's value model. We used Kirkwood's Strategic Decision Making: Multiobjective Decision Analysis with Spreadsheets (Kirkwood, 1997) as the source for this vocabulary.

There are several important considerations to keep in mind when developing a value hierarchy. These properties include completeness, nonredundancy, decomposability, operability and small size. A value hierarchy is complete when each tier of evaluation considerations adequately captures all concerns necessary to evaluate the overall objective of the decision. To be nonredundant, no evaluation considerations in the same hierarchy tier should overlap. The properties of completeness and nonredundancy combine to make the evaluation considerations in each tier of the hierarchy collectively exhaustive and mutually exclusive. This will be critical when the measures combine to reach an overall evaluation of each alternative.

Decomposability, or independence, is a property of the evaluation measures. The value model achieves decomposability when variations in the level of one measure do not impact the yield of any other measure. The following example illustrates lack of decomposability. "Suppose that a job seeker has an evaluation consideration 'economic issues,' and has proposed as lower-tier evaluation considerations for economic issues the following: 'salary,' 'pension benefits,' and 'medical coverage'" (Kirkwood, 1997: 18). These are not decomposable, because a very high salary could make medical coverage less of a concern and vice versa. Similarly, a high pension could make salary less of a concern. Such interdependency is problematic for combining the evaluation measures to determine overall alternative performance.

"An operable value hierarchy is one that is understandable for the persons who must use it" (Kirkwood, 1997: 18). Operability may be the most important quality of a value hierarchy. This is certainly true when the hierarchy must be meaningful to people with backgrounds different from those responsible for the hierarchy. Thus, when developing the hierarchy, it is important to consider the eventual audience. The small size quality is just what it infers. A smaller hierarchy is more desirable than a larger one. "A smaller hierarchy can be communicated more easily to interested parties and requires fewer resources to estimate the performance of alternatives with respect to the various evaluation measures" (Kirkwood, 1997: 18).

> The 'test of importance' (Keeney and Raiffa 1976, Section 2.3.2) provides a rough indication of whether to include a particular evaluation consideration in a value hierarchy. This test states that an evaluation consideration should be included in a value hierarchy only if possible variations among the alternatives with respect to the proposed evaluation consideration could change the preferred alternative. (Kirkwood, 1997: 19)

Keeping these qualities in mind while developing the value hierarchy will provide the most effective value hierarchy.

3.4 Identify Alternative Architectures

The process of identifying the alternative architectures employs some of the principles outlined in the system synthesis step of the systems engineering process. Before delving into the nuts and bolts of conceptual engineering (Sections 3.5 and 3.6), it is important to define an overall concept of the different alternatives. Thus, this step defines the different overall concepts that could be employed to solve the problem for our customer. Additionally, this step will define the employment strategy we use in each of the architectures. Our problem statement addresses two major challenges – upgrade of GPS payloads and repair of failed components. These two problems are likely to necessitate different requirements. Therefore, the alternative architectures

may provide a solution to one or both of the problems. We define these overall architecture issues as employment strategies.

Defining different architectures was the first step in suggesting solutions for the problem we identified. The following figure presents a detailed breakdown of the alternative generation steps.

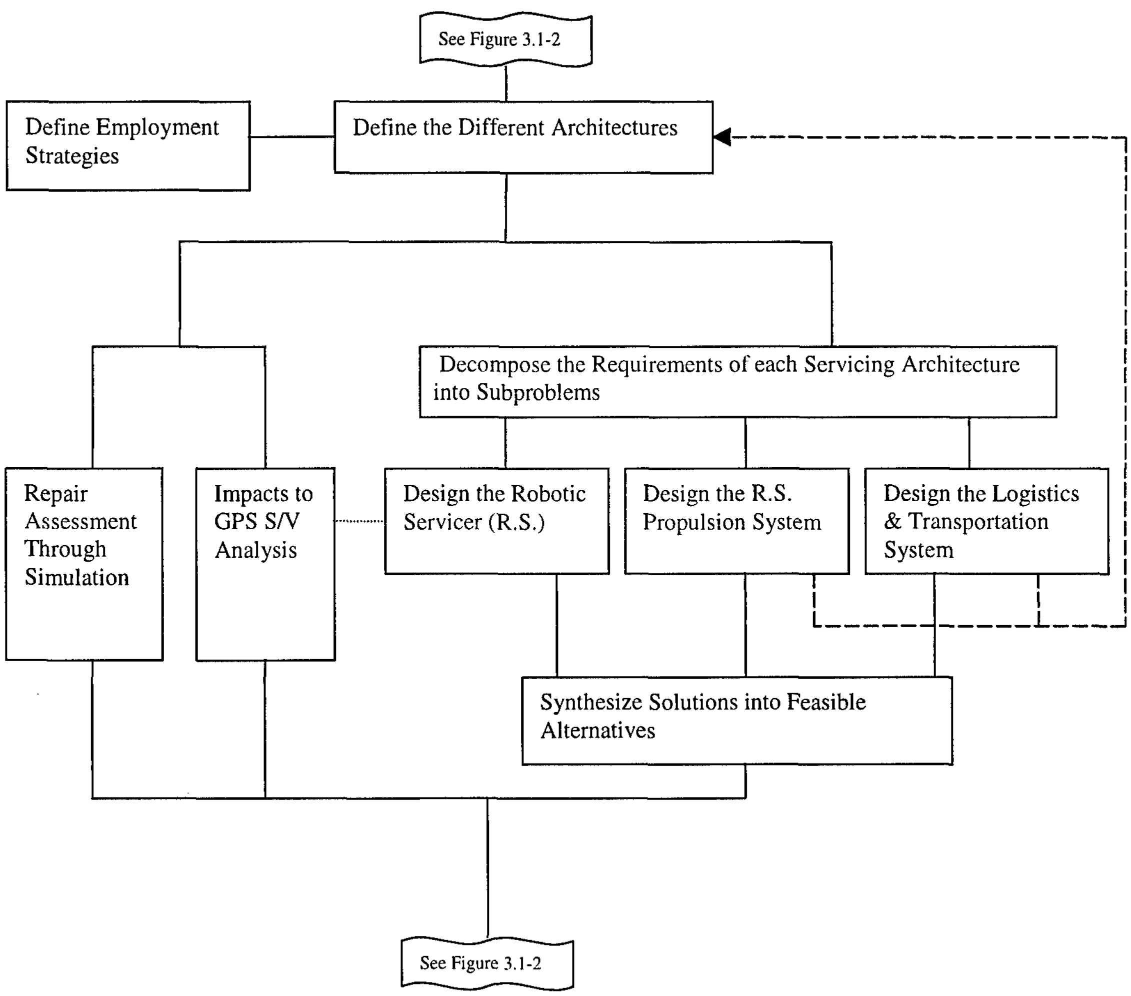

Figure 3.4-1. Alternative Generation Steps

3.5 Decompose into Subproblems and Design Solutions for Them

A major portion of our research focused on the design and modeling of different alternatives. This step encompassed the majority of the system synthesis and system analysis

steps in the SE process, and the decomposing and modeling steps of the DA process. Within this step we addressed both the Robotic Servicing System (RSS) and the GPS S/V. The RSS encompasses all the servicing functions and supporting transportation systems required to provide on-orbit servicing. The RSS decomposed into the following main subproblems: the robotic servicer, the robotic servicer's propulsion unit, and the logistics and transportation system.

In addition to analyzing the RSS, we analyzed the impacts to the GPS satellite vehicles (S/Vs) with two other studies. The first study was to characterize the extension in average GPS S/V life due to repair. We accomplished this using simulation. The second study was to assess the cost and mass impacts necessary to make the GPS S/V's serviceable. This is being performed in a companion study by Aerospace Corporation under guidance of the GPS Program Office.

3.6 Synthesize Subproblem Solutions into Alternative Solutions

"There is nothing so dangerous as a problem with only one solution" (Kramer, 1998). The purpose of this study is to see if there are any servicing architectures that would solve GPS's problems. As outlined in the last step, there are many different solutions to each subproblem of the servicing system. This step synthesizes these individual parts into a large set of alternatives that capture a broad spectrum of different solutions. It was beyond our scope to enumerate all possible alternatives. Accordingly, this synthesizing step will generate a diverse yet logical set of different alternative architectures. The alternatives will be generated from a common framework. The framework will be defined in Section 4.5.2.

3.7 Assess Value Functions

Below is a graphical elaboration of the value model elicitation portion of Figure 3.1-2. The only expansion from the process overview is in the value function assessment step. An explanation of the components follows the figure.

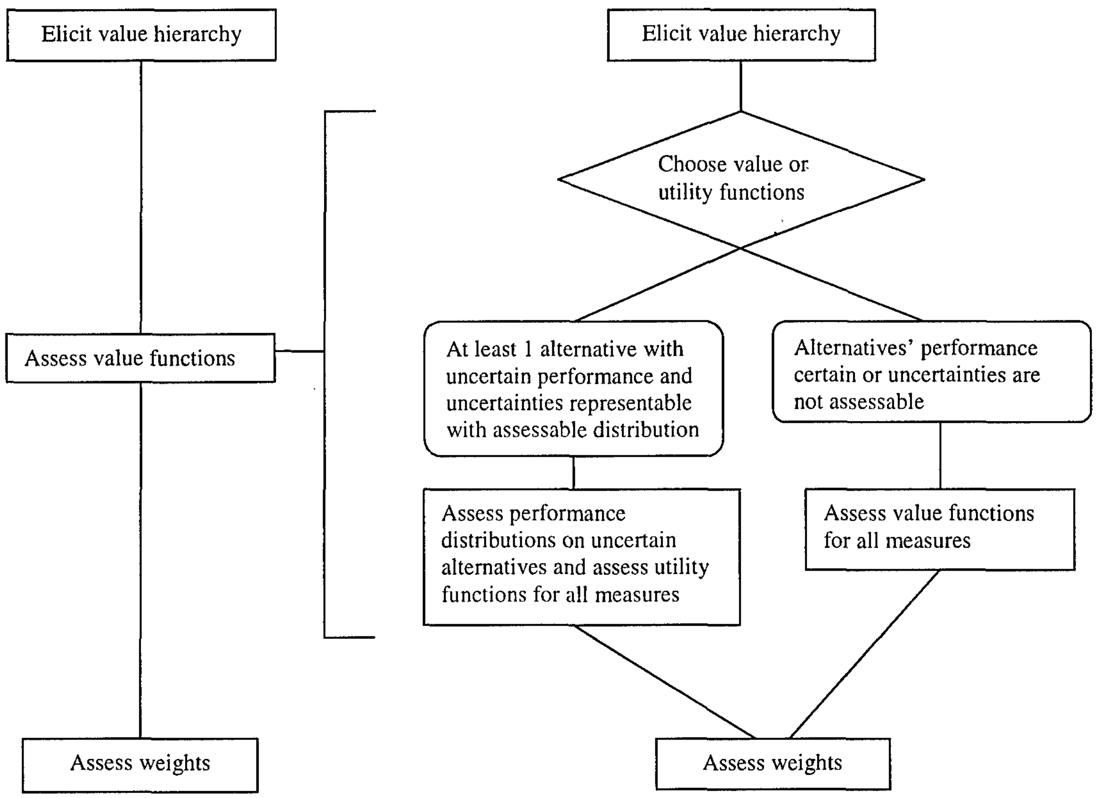

Figure 3.7-1. Value Model Elicitation Steps

With the value hierarchy and alternatives complete, the next step is to assess functions for each of the measures.

The general method of assessment for the measures involves some important decisions on the part of the analyst. First, it is important to decide how to incorporate the inherent uncertainty into the analysis. The uncertainty stems from the nature of the research. We were evaluating alternatives that existed only on paper. The performance of these alternatives was unknown, and the scores we used were theoretical. Thus, it was important to account for potential variation in

the actual performance of each alternative. One method of dealing with uncertainty is to use probability distributions to represent the major sources of performance variation in the alternatives. The scores then translate into utility through the use of utility curves. Finally, the overall utility for an alternative would come from expected utility calculations. In this analysis, however, due to the theoretical nature of the alternatives, we could not determine credible performance distributions. The method we chose was to deterministically represent the performance of each alternative. Then, we used sensitivity analysis to better understand the impact of this representation. This method translates scores into values using value functions, and the measures are combined using a weighted sum. This procedure is termed a multiobjective value analysis (Kirkwood, 1997: 53).

It is important to have both the hierarchy and the alternatives in place at this point. The hierarchy includes the measures, so the analyst knows what to assess. The alternatives provide important information that may impact the measures' ranges or weights. If the decision-maker assesses a range for a particular measure that excludes many or all of the alternatives' scores, it would be beneficial to understand why the decision-maker's expectations differed so significantly from the performance of the alternatives. It may be that we have improperly defined the measure. When it comes time to weight the measures, it is necessary to consider the range of the alternatives' scores. If the scores are primarily over a small range on the measure, a low weight will nearly eliminate the impact of that measure on the overall assessment. A large weight will cluster the overall values of the alternatives. A large weight for a measure on which the alternatives vary greatly in their scores will overshadow the impacts of the other measures. It is still important to perform the initial assessment of each measure without sharing the performance of the alternatives with the decision-maker. This sort of blind assessment helps to

reduce biasing the results towards the decision-maker's favorite alternative. However, it is important for the assessor to consider these interactions before moving to the next measure.

The end result of the value model is a rank ordering of the alternatives according to their overall score. See Chapters 4 and 9 of Kirkwood for value function assessment techniques. See Chapters 6 and 9 for utility function assessment techniques. The Midvalue Splitting Technique from Section 9.2 was appropriate for our purposes. To present this technique, the following definitions from Kirkwood are useful.

> **Definition 9.6. Strategic Equivalence:** Two value functions are strategically equivalent if they give the same rank ordering for any set of alternatives. (A rank ordering of a set of alternatives is a list of the alternatives in decreasing order of preference. That is, the first alternative in the list is more preferred than all the rest, the second is more preferred than all the others except the first, and so forth.) (Kirkwood, 1997: 229)
>
> **Definition 9.8. Additive Value Function:** Value function $v(x)$ is called an additive value function if it is strategically equivalent to a value function of the form
>
> $$v(x) = \sum_{i-1}^{n} \lambda_i v_i(x_i)$$
>
> For some functions $v_i(x_i)$ and constants λ_i. (Kirkwood, 1997: 230)
>
> **Definition 9.13. Midvalue:** When an additive value function is valid, the midvalue of an interval [x_i', x_i''] is the level x_i^m such that, starting from a specified level of another attribute, the decision maker would give up the same amount of that other attribute to improve x_i from x_i' to x_i^m as to improve x_i from x_i^m to x_i''. (Note that this definition implicitly assumes that preferences are monotonically increasing over x_i. If preferences are monotonically decreasing, then the same amount would be given up to improve x_i from x_i'' to x_i^m as from x_i^m to x_i'.) (Kirkwood, 1997: 233)

The first step in the value function assessment method is to develop the value scale for the function. The scale of value will go from 0 to some number, and it is important that all value functions are on the same scale. Common scales are 0 to 1, 0 to 10 and 0 to 100. The next step is to assess the practical maximum score and practical minimum score. The practical minimum is the minimum score below which there is no further decrease in value (in the monotonically increasing case) or increase in value (in the monotonically decreasing case). The practical

maximum is the maximum score above which there is no further increase in value (in the monotonically increasing case) or decrease in value (in the monotonically decreasing case). The next step was to assess a midvalue between the minimum and maximum. If the decision-maker or analyst sees the need for further resolution, they can assess additional midvalues between the scores they already assessed. For example, after assessing the midvalue for the minimum and maximum, the decision-maker and analyst could assess a midvalue between the minimum and the previous midvalue.

The method for assessing utility functions varies from the value function procedure. Utility functions may use exponential curves in conjunction with a value called the certainty equivalent.

> **Definition 9.26. Certainty Equivalent:** For a decision problem with a single attribute Z, the certainty equivalent for an uncertain alternative is the certain amount of Z that is equally preferred to the uncertain alternative. The certainty equivalent is also sometimes called the certain equivalent. (Kirkwood, 1997: 245)

It is possible to apply this generally to a decision problem with n attributes. Utility functions may also use piecewise linear functions. This research only required the use of value functions. See Kirkwood for further information regarding the use of utility.

3.8 Assess Weights

An evaluation measure's weight represents the overall increment in value the measure offers when ranging the score from its least preferred level to its most preferred level. The following observations help show this. In the monotonically increasing case, the practical minimum (least preferred level) gets a value of 0. The practical maximum (most preferred level), then, gets a value of the upper limit on the value scale. The value assignments reverse for the monotonically decreasing case. Thus, the range of scores for each measure translates into the

full range of values. Determining the appropriate weights for the measures is a relatively simple process. These steps are shown below.

1. Choose one baseline measure as a basis of comparison for the remaining measures. This is usually the measure that the decision-maker perceives to have the least impact or the greatest impact on the decision-making process.
2. Determine the impact relative to the baseline measure of varying each measure's score from the least preferred level to the most preferred level.
3. Set all the measure weights to sum to 1.
4. Solve for the individual measure weights.

This completes the assessment necessary to evaluate the alternatives.

3.9 Evaluate Alternatives

With the above tools in place, computationally evaluating the alternatives is a simple matter. The value functions, weights and alternative performance scores go into a spreadsheet. Using the capabilities of the spreadsheet, the analyst calculates the overall value for each alternative by a weighted sum and rank orders them. If the model intentionally did not include certain measures in the assessment and evaluation, it is now that the analyst can include them. For instance, the decision-maker may have held cost out of the model until the final evaluation. It would then be possible to compare overall value for each alternative versus that alternative's cost. This may add meaning to the assessment for the decision-maker. It is helpful at the beginning of the process to ascertain measures the decision-maker would like to see separately.

The mathematical results are the foundation for a complete evaluation of the alternatives. By carefully examining the model output, the analyst can draw conclusions that will provide insight for the decision-maker. That is the ultimate goal of this process. Our efforts have been successful if they provide the decision-maker and his staff with a tool that makes a useful contribution to the decision-making process.

3.10 Perform Sensitivity Analysis

The purpose of sensitivity analysis is to capture and quantify the robustness of the results. It is possible to perform sensitivity analysis on several aspects of the model. The weights are one candidate. Analysis of the weights provides users of the model an understanding of how sensitive the ranking of the alternatives is to the decision-maker's allocation of model emphasis. This is also an opportunity to evaluate performance uncertainties in the measures. If the analyst had to estimate data for a particular measure, sensitivity analysis provides a method to assess the impact of errors in the estimation. The sensitivities that arise during this step as well as those that do not arise offer valuable insights to the decision-maker.

3.11 Present Results

It is the analyst's job to find meaning in the model results. The analyst must draw insight from the numbers and convey that insight to the decision-maker. These results are a small piece of the picture to the decision-maker, but they can be a very useful and powerful piece. Its usefulness often depends on both the quality of the data and the quality of the presentation. The first thing to consider is the question that sparked the research. It is important for the presenter to stay focused on communicating an answer to that question, because that, most likely, is what the decision-maker is most interested in hearing. It is important for the presenter to consider the audience when developing the briefing. A technical briefing for a high-level manager may be the end of an analyst's contribution to a project if the manager cannot understand the information. On the other hand, an overview briefing may receive, at best, a lukewarm reception in the presence of a technical audience. Understanding the interests of the audience can make a significant difference in how that audience receives the information.

In addition to helping the originally intended customer, research results can often benefit others. Technical and academic journals provide a valuable forum for sharing professional experiences, results and discoveries. Symposiums and conferences are also forums for presenting information. Sharing valuable and useful information with others who can benefit from it should be a top priority of any researcher.

These methodology components all worked together to create a complete evaluation technique for this thesis effort. This technique made it possible to conduct detailed engineering analysis of alternatives while maintaining focus on the decision-maker's objectives. The outcome of this process provided an accurate and meaningful representation of the sponsor's needs. The following chapter details the results of applying this methodology.

IV Results and Analysis

4.1 Identify the Problem and the Decision

While Chapter One captured the overall problem and its scope, this section performs a more thorough definition of the problem. We needed a thorough and accurate problem description so that the generation and analysis of alternatives would stay appropriate for the needs of the user. For this reason, we completed the following section with our sponsor during the beginning of our research.

4.1.1 Problem 1: Long Delay for Implementation of New Capabilities

National requirements for payloads change quicker than the ability to deploy new hardware in a constellation of 24+ satellites. This problem includes the categories of preplanned product improvements and quick response to new threats. One illustration: suppose national needs require new civilian navigation capabilities to be deployed in 2 years. While the current generation of GPS satellites (Block IIA and Block IIR) have expected mean mission durations of 6 and 7.5 years respectively, the future generation of GPS satellites (Block IIF) will have a mean mission duration of 12.7 years. If an average IIF satellite lasts 12 years, it would require at least 12 years to deploy a new subsystem throughout the constellation. This time does not include design and acquisition of the new satellites.

The stakeholders for the navigation payload include the military, commercial, and civil users, the GPS Joint Program Office, the satellite contractors, and the GPS Control Segment. The stakeholders for the secondary payload are similar, including the national command authorities, agencies that process the NUDET (Nuclear Detection) information, and manufactures of the sensors. Stakeholders for any new payload could be any potential system users of GPS or its orbital altitude.

Future GPS payload requirements will drive future influential conditions for problem 1. For the navigation payload, this includes funding for a third L-band channel for civilian use, and funding for NAVWAR capabilities, which ensures continuous operation for the military user. For the NUDET payload, future conditions that influence on-orbit servicing are threat level of nuclear war, status of nuclear arms in other acknowledged nuclear powers, and proliferation of nuclear arms and testing in upstart nuclear countries.

4.1.2 Problem 2: Budgetary Constraints Require Innovative Ways to Reduce Cost While Increasing Capability

The GPS program must maintain its constellation by replacing satellites at the first actual or probable failure. As a result the Air Force has had to pay approximately $100 million every time a subsystem fails. If servicing reduces the cost of GPS replenishment, it could negate much the cost of the servicing infrastructure.

The stakeholders for problem 2 are similar to problem 1; the main difference is this problem affects the managers of the constellation more than the actual users. Therefore, to propose changes to the constellation, the GPS JPO would need to get buy-in from its funding sources more than its customers. In the Air Force these two organizations are generally different. Two important subjective considerations are the Air Force's policy on space logistics and the future GPS program budgets.

4.2 Elicit Value Hierarchy

The value hierarchy is an expression of the top-level objectives that govern an individual's or group's decision-making process. The following figure is the value hierarchy for a GPS decision regarding alternative constellation management architectures. The thesis team dealt directly with the GPS Space Segment office to develop this hierarchy.

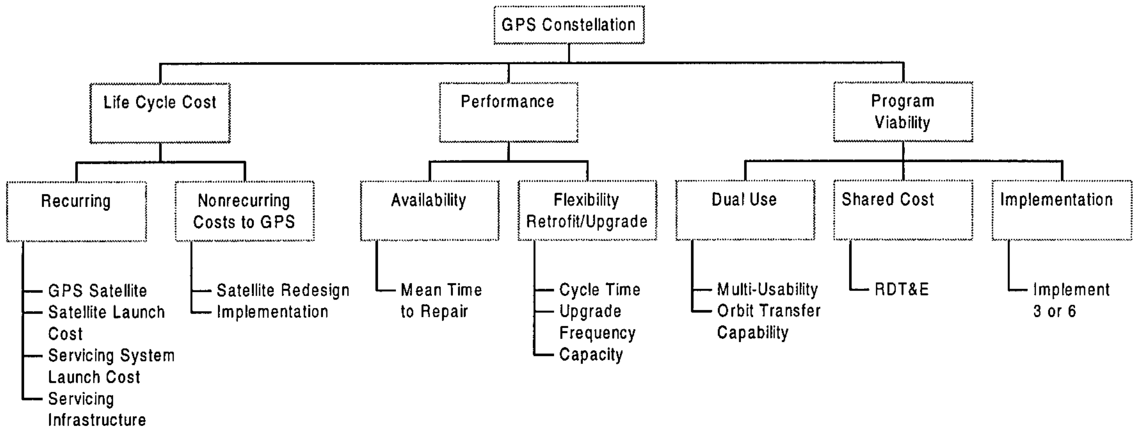

Figure 4.2-1. Value Hierarchy

The top of the GPS value hierarchy is GPS Constellation to reflect the general need to have the best GPS constellation possible. An equally appropriate top level would be "Choose the best possible constellation management alternative." This research focused on on-orbit robotic servicing alternatives. However, the value hierarchy and associated model could evaluate any constellation management alternative. The first-tier evaluation considerations represent the fundamental areas of concern that any alternative must address. These considerations are Life Cycle Cost, Performance, and Program Viability. Life Cycle Cost is an accounting of all costs to GPS for an alternative. Performance is the performance of the constellation when GPS implements a particular alternative. Program Viability reflects the likelihood an alternative would pass the scrutiny of approving bodies such as the Air Staff and Congress.

The second-tier evaluation considerations under Life Cycle Cost are Recurring and Nonrecurring Costs to GPS. Recurring is the recurring costs to GPS. The first evaluation measure for this is named GPS Satellite. This is the recurring hardware cost of the satellites. Satellite Launch Cost is for satellite launch costs, and Servicing System Launch Cost is for launches associated with the servicing system for which GPS is financially responsible. The fourth measure is Servicing Infrastructure. This accounts for the recurring costs of implementing

a particular alternative. The two measures for Nonrecurring Costs to GPS are Satellite Redesign and Implementation. Satellite redesign covers the costs for making the GPS satellite hardware serviceable. Implementation is the cost for changing the way GPS does its mission in accordance with a particular alternative.

The first-tier evaluation consideration of performance has two second-tier considerations. They are Availability and Flexibility. Availability is a consideration for repair alternatives. The measure for this is Mean Time to Repair, and the objective is to reduce this relative to the current constellation management alternative. Under the existing system, GPS managers can request a 60-day launch call when a satellite fails or is expected to fail. After launch, the new satellite takes 30 days to get into place, check out, and be considered operational. Thus, the objective on mean time to repair is for an alternative to be faster than 90 days. Flexibility Retrofit/Upgrade is a consideration for alternatives that can perform retrofit and upgrade. The measures are Cycle Time, Upgrade Frequency, Capacity, and Servicer Capability. Cycle Time is the time from launch of the first retrofit or upgrade to the time the upgrade reaches Full Operational Capability (FOC). FOC occurs when the modification is on 24 or more satellites. Upgrade Frequency is the number of times an alternative can upgrade a constellation during the constellation's life. Capacity is the mass of upgrade that the servicer can put on each satellite.

The Program Viability evaluation consideration has the measures Dual Use, Shared Cost, and Implementation. Dual Use reflects the usability of an alternative design by other satellite programs. The measures for it are Multi-Usability and Orbit Transfer Capability. Multi-Usability quantifies the capability of the servicer. Orbit Transfer Capability notes the maneuverability of the servicer for a particular alternative. The measure for Shared Cost is RDT&E, which is Research, Development, Test and Evaluation costs for an alternative. This

measure falls under program viability, because GPS is in a position to support the expenditure of RDT&E funds for a servicing alternative but would not be responsible for funding the program. Thus, this is a measure for viability, because the RDT&E estimate must be small enough for GPS to feel comfortable backing it. The second-tier evaluation consideration under program viability is Implementation. This captures the impact of an alternative that involves GPS transitioning to a 3-plane configuration. The measure Implement 3 or 6 reflects this. This completes the value hierarchy.

4.3 Identify Alternative Architectures

4.3.1 Process for Developing Architectures

Drawing on the background research we performed, and conforming to the realities of orbital dynamics and space systems, we outlined the different alternatives that seem feasible and economical. During the synthesis, we needed to conduct a fair amount of orbital dynamics analysis. This was an iterative process, and we will explain the evaluation process for orbital dynamics in Section 4.4.

4.3.2 Define the Employment Strategies (ES)

The first step in developing architectures was to establish clear design goals. While the value model provided measures to gauge which alternative Robotic Servicing System (RSS) best met the needs of GPS, the measures could not translate objectives into actual strategies for servicing by which we could design different alternatives. By defining how the GPS program will use the RSS, we gained those strategies. Those strategies will help us make assumptions regarding how we should structure each alternative.

4.3.2.1 Employment Strategy I: Service for Upgrade and Retrofit Only

The goal of this employment strategy was a low-cost RSS that could upgrade or retrofit in a scheduled manner. This strategy assumed the GPS satellite vehicle (S/V) was capable of upgrade or retrofit. Two of the key performance variables were the time and capacity for upgrading the constellation.

4.3.2.2 Employment Strategy II: Service for Scheduled Upgrade and Repair

The goal of this strategy was a low to medium cost RSS that could perform both upgrade and repair in a scheduled manner. The servicer would perform repair servicing when a subsystem failed down to single string. Not only must the GPS S/V be designed for upgrade and repair, but it must also be able to identify the failed component. Four key performance variables for the servicer were the nominal time for servicing of the constellation, the nominal time for servicing a selected satellite, the percentage of a GPS S/V that is repairable, and the capacity for upgrade and repair.

4.3.2.3 Employment Strategy III: Service for Upgrade and Quick Response Repair

The goal of this strategy was a medium to high cost RSS that could upgrade and maintain in a scheduled manner and quickly repair a satellite failure. This employment strategy would have some overlap with ES II. ES III might have the highest cost RSS, but it would also offer the most benefit. This makes the same assumptions of the GPS S/V as ES II. Not only were the four performance variables for ES II important, but the average time to respond to a failure was also important.

4.3.3 Terminology

Terminology needs to be explained in order that architectures have clear definations. The parking orbit is the orbit into which the launch vehicle (LV) would place the canisters. Canisters

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DESIGN AND ANALYSIS OF ON-ORBIT SERVICING ARCHITECTURES FOR THE GLOBAL POSITIONING SYSTEM CONSTELLATION

THESIS

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are described in Section 4.4.3. Parking orbits would only receive use if the canisters had propulsion units to maneuver into their target orbits. Otherwise, the LV would place the canisters directly into the target orbit. The target orbit is the final destination for the canisters. In the target orbit, the RS would rendezvous and on-load the required ORUs. The target orbit would either be the GPS orbits or the precessing depository orbit for Architectures "E" and "F".

A depot is the function of storing spare ORUs. The above mentioned depository orbit could be a depot or a way-point for ORUs already dedicated to at S/V. The depot could be a ground based system, which would launch ORUs a certain intervals. In addition, the depot could also be a space-based system with spare ORUs stored in space for quick response for failures.

4.3.4 Description of Alternative Architectures

4.3.4.1 "A" – Robotic Servicer (RS) in Each Orbital Plane: Short Term Upgrader

This architecture is the most simple concept. With a RS in each of the GPS orbital planes, we could eliminate the delay and cost of the RS making orbital plane changes. Each robotic servicer would upgrade the GPS satellite vehicles in its plane and then would undergo disposal. This option would be attractive if RS production cost is low.

4.3.4.2 "B" – RS in Each Orbit: Long Term Upgrader

This architecture was very similar to "A". The main difference was the RS would have a design life of 10 or 20 years, enabling it to make multiple upgrades. For the 2nd and following upgrades, the program managers would only have to launch the ORUs.

4.3.4.3 "C" – RS in Each Orbit: Upgrade and Semi-scheduled Repair

This architecture has the same robotic servicing system (RSS) as "A" or "B". The difference is this architecture involves both upgrade and repair of the GPS, whereas the earlier architectures only involved upgrade. Therefore, for the rest of the RSS analysis, only

Architectures A and B will be analyzed. Unlike architecture "D", this option would have the depot of spares on the ground.

This architecture, along with the following 3 architectures, is designed to offset the cost of the robotic servicing system by reducing GPS satellite costs. GPS satellite cost would be reduced through extension of GPS S/V life. The benefit of this extension would be determined through our simulation.

4.3.4.4 "D" – RS and Mini-depot in Every GPS Orbit

This architecture is similar to "C" except the depots are located in the GPS orbits. Thus, when a failure occurs, the RS picks up a replacement unit and fixes the failed satellite. The time to fix the failure is the time required for the RS to phase within the plane to the failed satellite and perform the service. This time would be much shorter than a 60-day launch call.

4.3.4.5 "E" – Precessing On-orbit Depot: Advanced Propulsion

The ORU depot would be in a precessing depository orbit that would align itself with the different GPS orbital planes at regular intervals of time. The RS would be stationed with the depot. At the correct lead time, the RS containing the necessary ORUs would make an orbital maneuver to rendezvous with the appropriate GPS plane and service the necessary GPS satellites in that plane. Once finished, the RS would perform the necessary maneuver to rendezvous again with the depot. In this architecture the RS would use an advanced low thrust, high I_{sp} propulsion system.

For upgrade missions, the RS could transport the ORUs from the depot, or their launch vehicle could deposit them directly into the GPS plane using the dispenser method. For a description of the dispenser method see Section 4.4.3.7. This study placed the canisters directly

in the GPS orbit. This help minimize the size of the already large RS propellant tank by reducing the requirements on the RS.

For the advanced propulsion, we chose solar thermal over electric propulsion for two reasons. First, although it would require multiple burns to complete the required change in orbital velocity, solar thermal performs the maneuver similar to an impulsive burn and, thus, can use the LEO to GPS transfer orbit as the precessing depository orbit. Electric propulsion requires a circular spiral transfer orbit, therefore the depository orbit would also have to be circular. The difference between these depository orbits is that the launch vehicles can place canisters directly into the elliptical orbit, but they would require an apogee kick stage for the circular orbit. The second reason for choosing solar thermal is that electric propulsion requires long transfer times between the depository and GPS orbits. This long time in the transfer orbit puts the RS's orbit behind the precessing orbit in longitude of ascending node. This would be hard to correct since the ability for electric propulsion to change longitude of ascending node is not much greater than the change in the precessing orbit due to the oblateness of earth.

4.3.4.6 "F" – Precessing On-orbit Depot: Chemical Propulsion

This is similar to "E" except that the RS would use chemical propulsion. To be able to rendezvous with multiple GPS orbital planes, the RS requires a re-supply of its propellant in the depository orbit. This re-supply could be concurrent with ORU re-supply or be a dedicated mission. An advantage this architecture would have over "C" is that many of the launch vehicle candidates could launch directly into the depository orbit. In "C" the ORU transport would require a propulsion system to get in the depository orbit.

4.3.4.7 "G" – Upgrader with Direct Plane Change Capability: Ion Propulsion

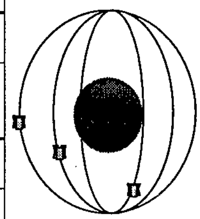
This architecture is similar to "A" in that it only performs upgrades. However, in this architecture there would be one RS and it would maneuver between planes. While reducing the number of robotic systems, this option requires a large propulsion system and additional mass for propellant. For this reason, this and the next architecture used advance propulsion systems with high I_{sp}'s (see spreadsheet #4). This requires less robotic servicers, but more time to cover the entire constellation (spreadsheet #4).

4.3.4.8 "H" – Upgrader with Direct Plane Change Capability: Solar Thermal Propulsion

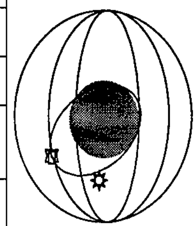
This was very similar to "G" except that it used solar thermal propulsion. While its thrust is miniscule compared to chemical propulsion, solar thermal had a tremendous amount of thrust compared to ion propulsion. This means solar thermal could complete upgrading the constellation in 11 months compared to 36 months for ion propulsion (see spreadsheet 4). The main disadvantage for this architecture is with the lower I_{sp} (approximately 800 seconds versus approximately 3100 seconds); it required much more propellant (see spreadsheet #4). The following table summarizes the architectures

Alternative	A	B	C	D	E	F	G	H
Architecture	R.S. in each orbit			RS & mini-depot in every orbit	One On-orbit Depot		Orbital Plane Change Upgrader	
	Short term upgrader	Long term upgrader	Upgrade and repair		Electric propulsion	Chemical propulsion	Electric Propulsion	Solar Thermal
Employment Strategy (E.S.)	E.S. I		E.S. II	E.S. III	E.S. II		E.S. I	
Life (yrs.) of R.S.	1 / 3	15		15	15	15	2	
# operational robotic servicers	3 (or 6)			3 (or 6)	1		1 (maybe 2)	
Depot	N/A		Ground	On-orbit	On-orbit		N/A	
Dedicated Launch Method	Multiple orbits (use the LEO dispenser method for both ORU's and R.S.)				One orbit for both ORU's and R.S.		Multiple for ORU, one for RS	
Piggyback beneficial?	Yes				No	Yes, into T.O. orbit	Yes, very much	
RS propulsion for plane changes	N/A				Med. Thrust	High Trust	Low	Med.
Launching RS with ORU's beneficial ?	Yes				No		No	
Refuel RS	No				Yes	Yes	Maybe	

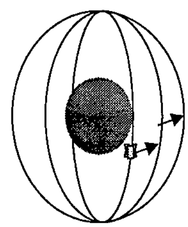
Alternatives A - D



Alternatives E , F



Alternatives G, H



Notes: Any of these alternatives can have 3 or 6 planes - for Alternative A and B the # of R.S.'s change
Low thrust: Electric propulsion (very high Isp)
Med. Thrust: Solar thermal propulsion (med. Isp, med. Thrust)
High thrust: Chemical propulsion (high thrust, low Isp)

Employment Strategy
(E.S.) I is to service for upgrade (and retrofit) only
E.S. II is to service for upgrade and scheduled repair
E.S. III is to upgrade and quick response repair

Figure 4.3-1 Summary of Different Architectures

4.4 Decompose into Systems and Design Solutions for Them

4.4.1 Decompose into Systems

On-orbit servicing involved two main systems. The first was the Robotic Servicing System (RSS) which included the transportation and robotic servicing elements (Sections 4.4.2 – 4.4.9). The second system was the user satellite that would receive service. In the case of on-orbit upgrade and repair, there were two elements of the user satellite that needed study. First, the benefits to the user satellite program for on-orbit servicing would need assessment. Upgrade benefits were straightforward and receive discussion in Sections 4.6 – 4.8. We used simulation to assess the repair benefits. We describe this portion of the evaluation in Section 4.4.10.

Second, there were cost and mass impacts to make the satellite serviceable. We addressed this with the help of a companion Aerospace Corporation study. A discussion of our interface with their study is in 4.4.11.

4.4.2 Robotic Servicing System Decomposition

In this step we have defined and investigated the separate components of the alternative architectures. By using the top down approach, we defined different overall systems functions and synthesized different feasible alternatives for each of the systems. As we mentioned before, the orbital architectures (Section 4.3) gave structure to the different concepts and we utilized them throughout the analysis. In fact, we added and changed the orbital architectures as we learned more about the system characteristics through the analysis of the following subsystems. The first main system was the Logistics and Transportation System (LTS). The LTS provides the function of transporting ORUs and Robotic Servicers (RS) to the orbits we designated. The system related to both the characteristics of the LTS and the rest of the robotic servicer is the Robotic Servicer Propulsion Subsystem (RSPS). Another system of the robotic servicer is the Robotic Manipulating & Bus Subsystems. The robotic manipulating subsystem was the payload of the RS. Its function was to provide the capability of adding or changing out ORUs on the GPS S/V's. The bus subsystems provided the power, communication, attitude control, and data processing for the robotic servicer. The robotic servicer propulsion subsystem maneuvered the RS between ORU canisters and GPS S/V's within an orbital plane, and possibly between GPS orbital planes. The below figure illustrates the decomposition just described.

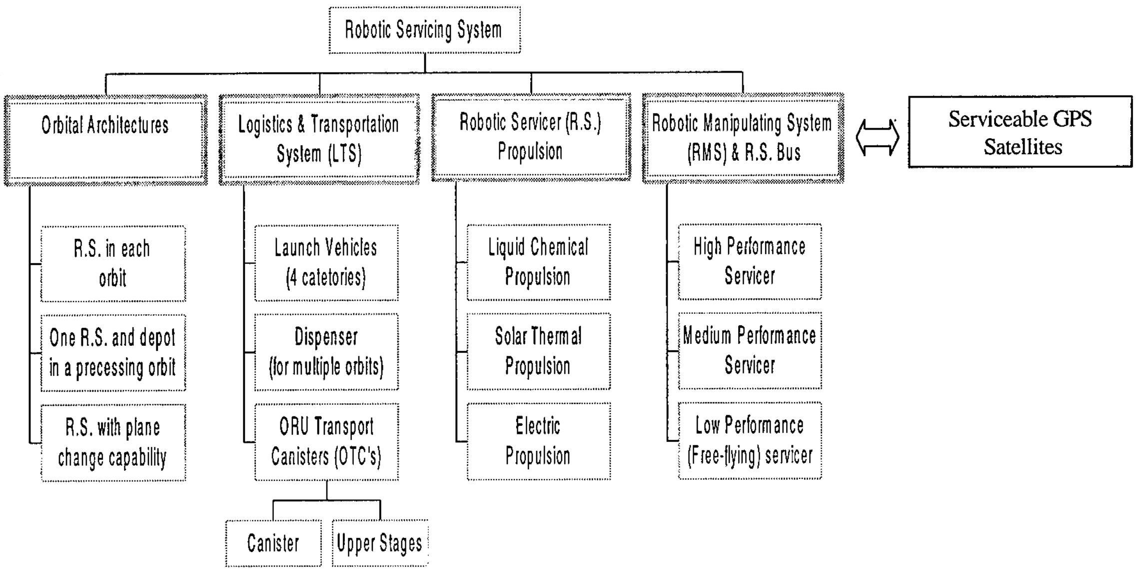


Figure 4.4-1. Decomposition of the Robotic Servicing System

4.4.3 Logistics and Transportation System (LTS) Analysis

While much of the focus of on-orbit servicing has been on the concepts for the manipulation of the satellite, it was important to realize that on-orbit servicing was also a transportation problem. The high cost of transporting systems into space emphasizes this fact. For many space programs, spacelift will be 50% of the mission cost. For example, a GPS IIF launch costs $50 million, while the satellite only costs $30 million (Wishner, 1999). By taking a systems approach to the overall on-orbit servicing concept, the study should highlight the important relationships between a space transportation system and on-orbit servicing (see Section 5.5). A complicating factor for designing a transportation system was the wide variety of transported mass requirements for different alternatives. Therefore, generating low cost transportation alternatives with a variety of capacities was our focus during the LTS design phase.

4.4.3.1 Orbital Design for Logistics and Transportation System

4.4.3.1.1 Spreadsheet #1: Insertion into LEO Parking Orbit

In this scenario, the LV launches the canister(s) into a circular low altitude, parking orbit. From the parking orbit the canister's propulsion system maneuvers the canisters into their respective target orbits. If the mission has more than one canister, the launch vehicle will have a dispenser like Iridium or Globalstar (Butris, 1998). The canisters will separate from the dispenser and stay in low earth orbit (LEO) until aligned with their target GPS orbit.
By launching into LEO, an opportunity arises for one launch vehicle insert canisters into to three or six different orbital planes. The effect of the earth's oblateness on the precession of an orbit plane is quite different between LEO and GPS orbits. Therefore, a LEO plane will align itself with all the different GPS orbital planes in approximately 20 days. Thus, we could launch canisters to all of the GPS planes on one launch vehicle. In essence, we let nature perform our plane changes for us instead of using costly propellant. Appendix A-1 describes the detailed development of spreadsheet #1.

Average costs of producing the LTS components depend on the total amount manufactured components. While the number of launch missions varied, two good numbers to assume were two or eight launches. Therefore Appendix A-2 shows the cost calculations for eight launch missions. Appendix A-3 shows the cost calculations for two launch missions.

To verify this spreadsheet's algorithms, we used the only mission to this orbit – GPS. Comparing the Delta II one canister column from the spread sheet with a GPS satellite:

Table 4.4-1. Verification of Delta II Payload Masses

	LEO	Transfer orbit	Final GPS orbit
Spread sheet	5092 kg	2139 kg	927 kg
GPS IIA (Wilson, 1994: 181)	N/A	1881 kg	930 kg

The transfer orbit mass is different between the spreadsheet calculations and the actual GPS IIA mission. The reason is that the spreadsheet has all the inclination change happening at GPS orbit, whereas the real mission puts the GPS S/V in a 37-degree transfer orbit.

4.4.3.1.2 Spreadsheet #2: Insertion into GPS Transfer Orbit

The development of this spreadsheet is very similar to spreadsheet #1. The main difference is the launch vehicle inserts its payload (the canister{s} and dispenser {if necessary}) directly into the transfer orbit. Thus the canisters need a propulsion subsystem only for the apogee kick maneuver. The main disadvantage is the precession rate in the transfer orbit is much less. Thus the orbital plane requires 300+ days versus 18-25 days to precess around to the different GPS orbits. However, if the RS has to transverse between planes or the transfer orbit is the depot orbit a slower precession rate results; however this should not be a large drawback.

Launch vehicles have different performances to different orbits. Thus, there is a difference in performance comparing a direct launch into a transfer orbit with a LEO launch with an upper stage. This creates a difference in final outputs between spreadsheet #1 and #2. Appendix B illustrates an example of spreadsheet #2.

4.4.3.2 Launch Vehicles

We chose four launch vehicles representing four different categories. With our goal to find the optimal overall architecture, the differences between competitive rockets within the same category was small compared to the differences between categories. There were four criteria for choosing the launch vehicles. They needed to be competitive in performance and

price. They needed to have a significant amount of data available. For our analysis 3 of the 4 LVs had their payload planner's guides (PPG) on the Internet. Next, the LVs needed to stay on the market for the next 10 - 15 years. Finally, they needed to already be either launching or planning to launch DOD payloads. The Kistler rocket was the only exception, because the DOD had not yet dedicated any payloads to reusable launch vehicles.

4.4.3.2.1 Intermediate Launcher: Evolved Expendable Launch Vehicle (EELV) Medium & the Medium + Class Boosters

The Medium + class EELV is a medium core booster with solid rocket motor (SRM) strap-on's added to the first stage. For this analysis, the (5,4) configuration of the Medium + booster was chosen. The (5,4) configuration has a 5.1 meter diameter fairing and 4 SRM strap-on's.

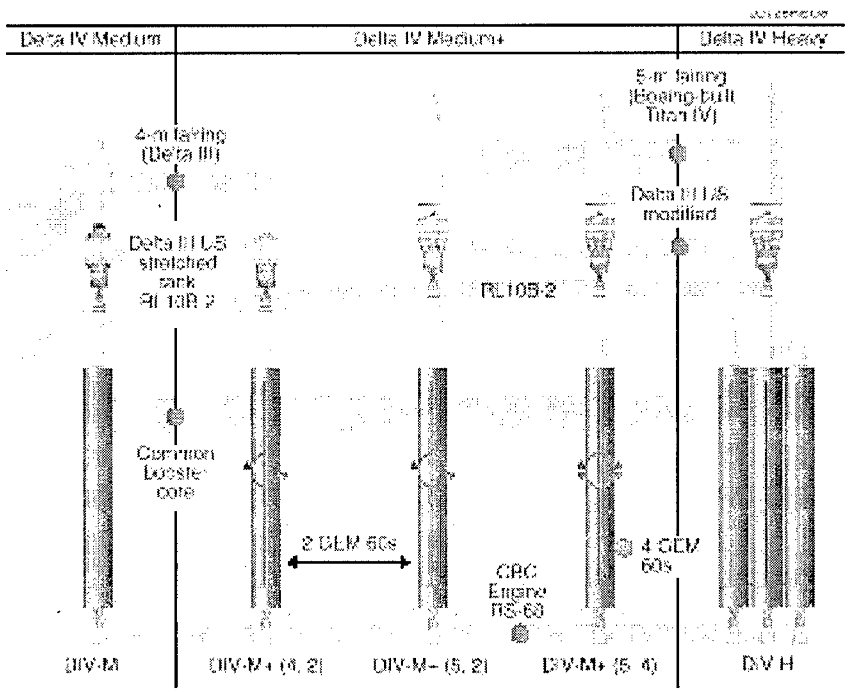


Figure 4.4-2. Delta IV Launch Vehicle

The LEO performance figures were in Figures 2-11 and 2-36 of the Delta IV Payload Planners Guide (PPG). The GPS Transfer Orbit (TO) performance numbers were in Figures 2-13 and 2-38 of the PPG.

With the EELV Program in the acquisition phase, cost data was proprietary. However, with the Air Force's EELV goal of 25-50% cost reduction below current DOD launchers (Delta II, Atlas II, and Titan IV), we represented the cost of the rocket in the current category using a 25% reduction. Later, we were able to verify this when the EELV program office informing us that the quotable EELV medium price is $75 million (Joyce, 1998). However, this study used the calculated $73 million so it was consistent with the $95 million value for the M+ cost. The equations are shown below.

EELV Medium → Atlas IIA

75% of $97 M = $73 Million* Equation 1

EELV M+ → Atlas IIAS

75% of $126 M = $95 Million* Equation 2

* (Middle value of range quoted in NASA cost page ["Cost Estimating", 1998: 2])

4.4.3.2.2 Medium Launcher: Delta II

With the elimination of the small class of EELV, the Air Force is left without a future booster in this category. However, there are three reasons why using the Delta II has merit in representing this launch category. First, the Delta II is the mainstay of American rockets in this category with very good reliability, low cost, and the Air Force (GPS in particular) has much experience with this rocket. Second, with Boeing canceling development of EELV small, the Delta II is their only booster in this category and they will keep launching these boosters as long as there is demand (Anselmo, 1998: 71). Finally, Capt. Karuntzos, from the Air Force's medium

launch vehicle program office, said the Air Force has not addressed the loss of Air Force available boosters in this category. His option was that if the user showed justifiable reasons for Delta II launches, they could contract for them (Karuntzos, 1998). Another method could be for the DOD to contract the launch service commercially. This method also would make the Delta II available.

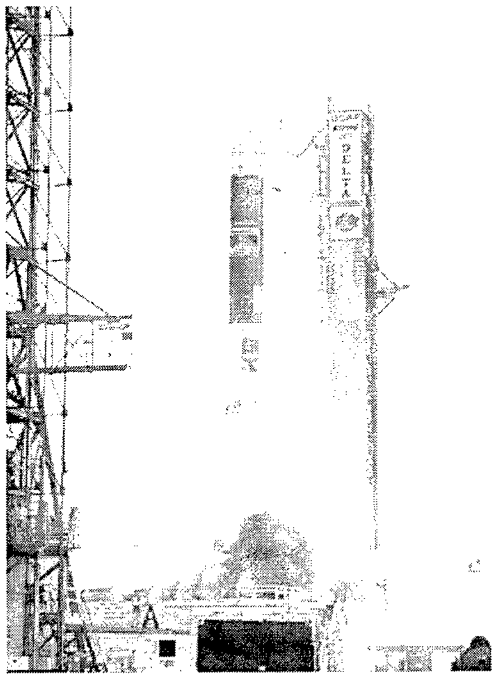


Figure 4.4-3. Delta II Launch

To find the performance of a Delta II to LEO (Delta Model 7920), we took the average between the 5139 kg value in Boeing's payload planner's guide (Delta II PPG, Fig. 2-6) and analysis by the Teal Group (5045kg) (Iannotta, 1998: 36,37). The performance numbers to GPS Transfer Orbit (Delta Model 7925) are very accurate because of the 28 GPS launches. With the Delta II being the only mature launch vehicle in this analysis, we took the bottom end of the cost range in NASA's cost page.

4.4.3.2.3 Small Launcher: Taurus XL (4 Stage Version)

Orbital Science has 3 versions of the Taurus launch vehicle: Standard, the XL, and XLS. The standard is operational, the XL is in development, and the XLS is being studied. The Standard had little lift capability for the missions we are studying. With the launch vehicle market in continual change, it seemed too risky to include a commercial booster that is only being studied. Thus, the Taurus XL was a logical choice for this category.

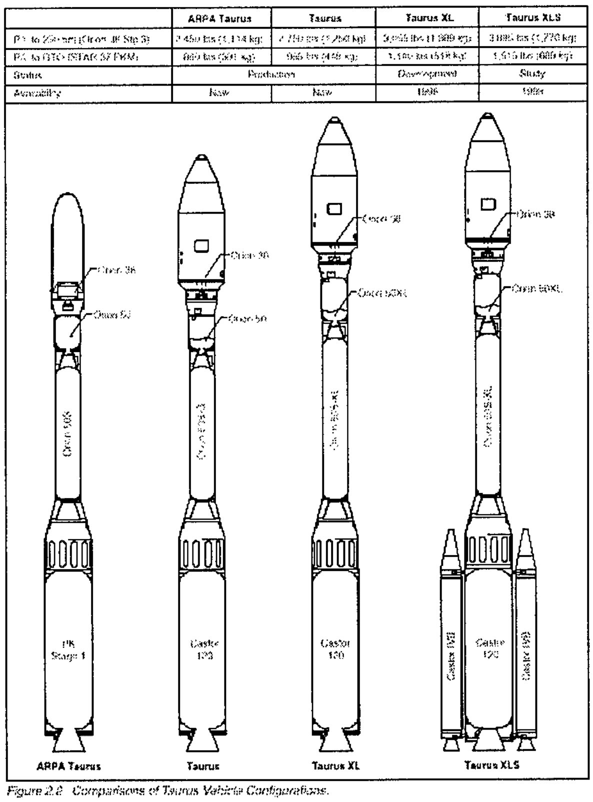


Figure 4.4-4. Taurus Launchers

To calculate LEO performance, we averaged the value from Teal's report and Taurus' PPG (p. 2-3). The payload planner's guide had no data for lift capability to GPS T.O. Therefore, we used the only available data – the Geosynchronous Transfer Orbit performance (the average value from Teal and Taurus's PPG is 557kg). Then we added 17% lift capability to get Taurus' performance to GPS transfer orbit (GTO performance is 85% of GPS TO performance). There

are two reasons for using this increase. First, the required change in velocity (Delta V) from LEO to GPS T.O. is 75% of that for GTO. The Delta V for GPS TO is 2.513 km / sec^2. The Delta V for GTO is 1.881 km / sec^2. Second, comparing launch vehicle performances between the two orbits also supports adding lift capability. The Delta IV lift performance to GTO is 88% of GPS TO (3900 kg verses 4420 kg). The Delta III lift performance to GOS is 86% of GPS TO (3800 kg versus 4400 kg). The Delta II lift performance to GTO is 87% of GPS TO (1819 versus 2040). The cost value for the Taurus is the average value from NASA's cost web page (Cost Estimating, 1998: 2).

4.4.3.2.4 Reusable Launcher: Kistler K-1 Reusable Rocket

There are many reusable launch vehicle efforts, and trying to analyze them all would be beyond the scope of this analysis. Thus, we chose one that is as far as any in the development cycle and has the financial backing to be a viable contender in the launch vehicle market. This year, Kistler is finishing its detailed design, production and testing of most of the K-1 components. The K-1 is planned to be operational by the year 2000. Also, the project is well-funded (approx. $200 million) and has been awarded over $100 million in launch contracts with Space Systems / Loral ("Kistler Development Schedule", 1998:2).

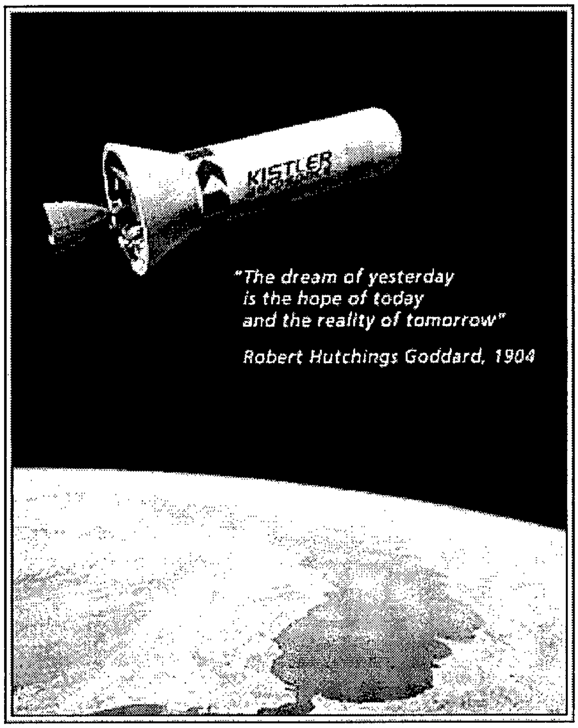


Figure 4.4-5. Kistler K-1 Rocket

To calculate the LEO performance of the K-1, we took the average from the values stated on Kistler's webpage and Teal's analysis. Being a future commercial endeavor it was hard to find accurate cost data. we used the average between the Kister's planned fee of $18 million (Proctor, 1997: 53) and a $48 million value obtained by Paul Yuhas (Yuhas, 1998).

4.4.3.3 Piggybacking

4.4.3.3.1 Concept

Since no amount of servicing can extend a satellite's life indefinitely, there will always be launches of new GPS S/V's. Future GPS S/V's will be launched on a medium class EELV. The EELV concept is that all DOD satellites will be launched on a common set of launch vehicles. Thus, the EELV medium is not designed optimally for launching GPS S/V's. In fact, there is a significant amount of margin between launch performance of the medium class EELV

and GPS IIF's mass. The Robotic Servicing System could use this margin for a "free ride" to go to the exact orbit we need (i.e. the GPS transfer or operational orbit).

4.4.3.3.2 Destination

The different RSS architectures have one of two different target orbits. Most alternatives need the RS or OTC's deposited into the GPS operational orbit. However, the precessing depository orbit alternative needs the payloads to be deposited into the transfer orbit.

4.4.3.3.3 Analysis

Since the IIF design is continually changing, a specific total mass is not available. However, the most conclusive requirement for GPS IIF is its launch weight requirement given in the EELV Request for Proposal (RFP). The RFP requires the EELV to place the 8175 lbs. (3716 kg) into GPS transfer orbit (EELV RFP, 1998:7). The Boeing medium class EELV (EELV IV PPG, 1998) can place 4,420 kg into GPS transfer orbit. This will enable us to add a piggyback OTC with a mass of 704 kg. With a canister structural ratio of .08 (see 4.4.3.5), the piggyback OTC can deliver 648 kg of ORUs to GPS Transfer orbit. With a solid upper stage the OTC can deliver 305 kg to the GPS operational orbit (see spreadsheet #2).

For a low cost method of delivering a large mass of ORUs to orbit, the GPS IIF S/V could be launched on a Medium + (5,4) launcher which would place 1,230 kg into the GPS operational orbit.

4.4.3.4 Orbital Transport Canister (OTC)

An ORU supply mission is composed of one to six OTC's depending on the number of GPS orbital planes that need servicing. The OTC is composed of two subsystems: the canister containing the ORUs, and the upper-stage(s) that transport the OTC to its target orbit.

4.4.3.5 Canisters

The function of the canister is to house the ORUs during the transport from earth to RS. The concept would resemble a cabinet full of ORUs. Since the canister is mostly just a simple mechanical structure, we chose the structural ratio of 8% (Larson and Wertz, 1992: 300, 321).

To calculate cost we chose the SMAD cost estimation relation for a structural subsystem. With little of the complexity or thermal requirements of a satellite's structural subsystem, we chose a RDT&E factor of 0.5. With this factor a 44 kg empty weight canister would cost $4.5 million. This canister would be used on the EELV medium launch vehicle.

4.4.3.6 Upper Stages

4.4.3.6.1 Solid Rockets

The Isp and structural ratios for the solid rocket motors were taken from Thiokol's Star family of motors. Using this family was advantageous because it had over twenty different sizes. This diversity enabled us to use many of the current Star motors for the transfer orbit missions outlined in spreadsheet #1 and #2. For example, the first mission on spreadsheet #1 was launching 3 canisters on an EELV medium class booster. Launch mass was 7,940 kg with a dispenser mass of 794 kg. The mass calculation of each of the three perigee kick motors was:

(launch mass – dispenser mass – payload mass) divided by three

$$(7{,}940 - 794 - 3{,}072) / 3 = 1358 \text{ kg}) \qquad \text{Equation 3}$$

This mass was close to the Star 31 with a total mass of 1393 kg. The mass for each of the apogee kick motors would be 480 kg, which was close to the Star 30BP with a mass of 505 kg.

We calculated the cost of upper stages using the Cost Estimation Relation (CER) table equation from Table 20-5 in SMAD (727). Those were in 1992 dollars, and a 1.175 inflation

factor was necessary to convert them into 1997 dollars (Larson and Wertz, 1992: 721). The spin stabilized Apogee Kick Motor CER equations were:

$490 + .0518*X^{1.0}$ *($K) for RDT&E* Equation 4

Unless noted otherwise the X value is the dry weight of the subsystem (in kg). This RDT&E cost was then multiplied by the development heritage factor. Since all our solid motors were existing, this multiplicative factor was 0.2 (Larson and Wertz, 1992: 728).

$0+ 58*X^{.72}$ *($1,000's) for first unit cost (FC)* Equation 5

The X value was total impulse (kg*s) where we converted the mass of the stage in kilograms to impulse using an average ratio for Thiokol motors (Wilson, 1994: 287,88) of 2.65 kNs / kg. We converted units by kNs = 1000 ($kg*m/s^2*s$) / 9.8 m/s^2 = 102 kg*s. However, the X value range for the FC equation was 8 to 57 kg, while our upperstages were much larger. Thus, we scaled the magnitude of this equation to fit a Delta II 3rd stage (Star 48) with a cost of $5 million ("Cost Estimating", 1998: 2) and a mass of 2137 kg. Thus, the new equation was:

$FC = 0 + 20*X^{.72}$ Equation 6

Additional propulsion system costs are calculated using the learning curve technique (Larson and Wertz, 1992: 734) with the following equations.

$Production\ cost = FC*L$

$L = N^B$

$B = 1 - \ln(1/S) / \ln 2$ Equation 7

The variables are defined as follows: FC is first unit cost, L is the learning curve factor, N is the number of units, B is the learning curve exponent, and S is the learning curve slope. We chose S = 90% based on recommendations in SMAD (735).

4.4.3.6.2 Liquid Rocket Engines & Solar Thermal

Both of these types of upper stages would be built for this specific mission. For a liquid rocket engine, we used an average value of Isp and structural mass ratio for nitrogen textroxide (N_2O_4) and monomethyl hydrazine (MMH). We chose this fuel and oxidizer combination because they have good performance while also being storable. For solar thermal propulsion, we used the Isp's and structural mass ratios given for the only designed solar thermal transfer vehicle, AFRL's SOTV.

We calculated cost of the liquid rocket engine upper stages using the CER table in SMAD. The 3-Axis Stabilized equations were:

$0 + .0156 * X^{1.0}$ *for RDT&E*

$0 + .0052 * X^{1.0}$ *for FC* Equation 8

The X value was total impulse (kg*sec). We used a multiplicative factor of .5 (moderate modification) for the RDT&E of the liquid rocket motors. In addition, we used the same learn curve formulas as the solid rocket motor calculations.

There is not much cost data on solar thermal rocket engines, so we used the only data available – the Boeing SOTV's brochure. They state SOTV's advantage was with a single propellant and a simple design will reduce stage cost by 30%. To stay conservative, we used a reduction factor of 15%. We included this reduction in the production cost, but increased the RDT&E cost (via the multiplicative factor) because this was a new design.

$0 + .0156 * X^{1.0}$ *for RDT&E*

$0 + .0052 * X^{1.0}$ *for FC* Equation 9

The multiplicative factor for the RDT&E was 1.3 because it was a new design with advanced technology. The multiplicative factor for the FC was 0.85 (Solar Orbit Transfer Vehicle Brochure, 1998).

4.4.3.7 Dispensers

4.4.3.7.1 Research

Since dispensing multiple satellites from one launch vehicle is a relatively new concept, there was no published data other than pictures in *Aviation Week & Space Technology.* All data for this system was from a phone interview with Mr. George Butris, a Boeing engineer who has worked on the Iridium and Globalstar programs (Butris, 1998).

4.4.3.7.2 Concept

Space dispensers can get very elaborate. An example of an elaborate dispenser is the MX dispenser, which repositions itself after each warhead release. However, the ones used in the Iridium and Globalstar programs were simple mechanical structures with some timing and latching mechanisms. In Iridium, the satellites were placed together on top of a platform dispenser. This configuration was like placing five glasses on top of a dish plate. Thus, each satellite was designed to receive the launch loads through the length of its structure. In Globalstar, the satellites were attached to a center mounted post. The advantage to this was the satellites had multiple attachment points to distribute the launch loads.

4.4.3.7.3 Mass Ratios

With Iridium's smaller design, the dispenser was approximately 5% of the total payload weight. With a much larger post design, Globalstar's dispenser was approximately 14% of the total payload weight. For our concept, whether launching into LEO or GPS transfer orbit, the dispenser would always be launching payloads that had an upper stage. Thus, both the RS and

canisters would have to be designed to withstand the longitudinal loads of an apogee and probably a perigee boost burn. Since they are designed in this manner they would tend to be in a configuration that supports bottom attachments like Iridium. With Iridium's platform dispenser has a lower mass ratio, I chose to baseline that concept. While Iridium's mass ratio was 5%, Mr. Butris recommended using 7% ratio for a generic platform dispenser (Butris, 1998). He did not feel the number of canisters would change the dispenser ratio. However, because of the increased complexity, we made the 6 canister dispenser ratio 8% versus 7% for the 3 canister dispenser.

4.4.3.7.4 Cost

With dispenser launching methods being a new commercial endeavor, Boeing was not willing to release any cost data. However, Mr. Butris did give strong credence to using SMAD's Cost Estimation Relation (CER) equations. He said the dispenser consisted of approximately 23 kg of timing and releasing mechanisms. The closest cost component in SMAD was Tracking Telemetry & Control (TT&C), and so we used TT&C CER equations. Notice we assume this doesn't change since size of the dispenser should have little bearing on the timing mechanisms. He said the rest of the structure was basically a mechanical structure for which SMAD has CER equations (Butris, 1998).

The 23 kg of timing and releasing mechanism correlated to:

$RDT\&E\ cost = 1955 + 199*(23) = \$\,6{,}532{,}000$

$\$\,6{,}532{,}000 * .1\ (existing\ design) = \$\,653{,}000$

$FU\ cost = 93 + 163*(23)^{.93} = \$\,3{,}103{,}000$ Equation 10

Average cost over 5 missions:

$B = 1\text{-}(\ ln(100\% / 90\%) / ln\ 2 = .848$

$L = 5^{.848} = 3.91$

$Average = \$3,103,000 * 3.91/5 = \$\ 2,429,000$ Equation 11

For the rest of the structure, we used the SMAD's structural CER's:

$RDT\&E\ cost = 2640 + 416 * X^{.96}\ (\$1,000s)$

$FU\ cost = 0 + 185 * X^{.77}\ (\$1000s)$

$Average\ cost = FU * 3.91/5\ (see\ above)$ Equation 12

The RDT&E cost had a multiplicative factor of 0.1 because it was an existing design. Since the platform structure is simpler than a satellite design, we scaled the production CER by 0.5.

4.4.4 RS Propulsion

4.4.4.1 Mass Ratio Analysis

In the LTS analysis, initial mass was an input variable (via lift capability of the booster) and final mass was an output. Thus, the mass structural ratios for the canister's propulsion system were based on initial mass. The structural ratio was ms / mo. For the RS propulsion system, final mass was an input variable and thus the mass ratios were based on final mass. The structural ratio was ms / mf.

The liquid propulsion system we considered was the in-plane phasing maneuvering system. Liquid propulsion between planes was only used in Architecture F, and was analyzed separately. Since the liquid propulsion structural ratio was based on empirical trends in current systems, the methodology used to size the propulsion system was important. In the canister propulsion system, the mission requirement (Delta V > 2 km/s) was similar to orbital transfer vehicles, and so empirical data from those systems was appropriate. Here, the requirement was much smaller (Delta V < .03 km/s for 12 day phasing time). Thus, the thrust to weight ratio was

smaller. This corresponded to a smaller rocket engine. Comparing the two systems we found the RS propulsion engine could be 1.5% of the canister's propulsion system (.03 km/s / 2 km/s). The propellant tank was sized according to amount of propellant, which was based on required total Delta V (.03 km/burn * 20 maneuvers [from Alternative B] * 2 burns/maneuver = 1.2 km/s). Combining the two effects, the structural ratio could be reduced to 35% of the previous LTS value. Using the same IABS numbers used in the LTS analysis, the liquid propulsion system's structural ratio became:

*ms / mf = 19.2% (227 / 1180) * 35% (sizing reduction factor) = 7%* Equation 13

Since a solar thermal propulsion system was low thrust, sizing was not as straight forward as liquid propulsion. In addition, solar thermal propulsion would be used more frequently for orbital plane change maneuvers. With the limited amount of theoretical data for solar thermal, we used the SOTV thruster and only sized the propellant tank. The SOTV engine minus the propellant tank and bus subsystems had a mass of 129 kg, and a thrust of 34.6 Newtons (Dornheim, 1998:76,77). The propellant tank took a larger role in sizing than an altitude boost mission like SOTV. The reason for this was that the RS has to carry the propellant for 30 phasing maneuvers and 5 plane changes, which has a total Delta V of approximately 10 km/s. The 30 phasing maneuvers stem from 5 phasing maneuvers per plane times the six planes in the constellation. The five phasing maneuvers result from rendezvousing with ORU canister in the orbital plane and then the corresponding four GPS S/V's (see spreadsheet #3 write up).

To size the propellant tank, propellant mass became both the design input and output variable. To calculate the amount of propellant needed for a maneuver, we need to know the total mass, which was dependent on the amount of propellant. To find a solution to this iterative process, we guessed at total propellant mass (top of page 2, spreadsheet 3) and compared it to the

real propellant mass (bottom of page 2, spreadsheet 4). If the two values were within 20% of each other we stopped. Otherwise, we inserted the real propellant mass as the new input and compared again.

Empirical data for hydrogen storage tanks varied dramatically. The small test version of the SOTV had a tank mass of 51 kg with 83 kg of propellant with a tank to propellant mass ratio of 61%. The Space Shuttle external tank had a mass of 35,000 kg with 703,000 kg of propellant (Wiesel, 1997: 206), with a mass ratio of 5.0%. With this much variation we used 10% based on SMAD's 5 – 15% range (660).

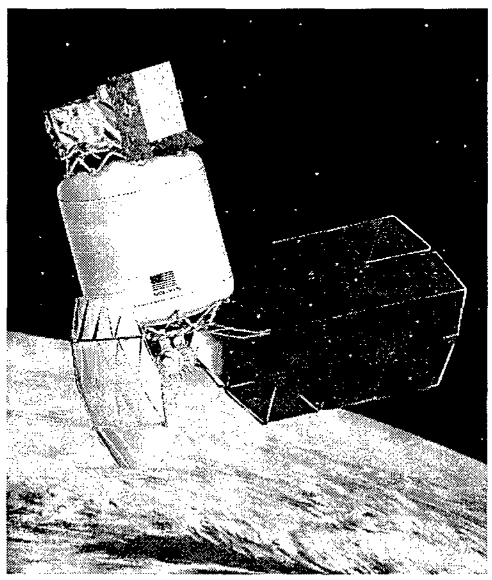


Figure 4.4-6. Solar Thermal Propulsion Conceptualization

Solar thermal tank mass played a significant role in total mass of the RS. In the worse case scenario (RMS and RS bus = 350 kg, mass of ORUs = 300 kg, Architecture E) the 10% tank version required a RS initial mass of 5,000 kg. The 15% tank version required an initial mass of 6,200 kg. Thus, Architecture E did not employ the largest RS or ORU size. In addition, we used the 10% tank version, but if Architecture E became attractive after the evaluation, this assumption should be studied further.

Electric propulsion offers tremendous savings in propellant mass due to its incredible Isp. We chose xenon ion propulsion for two reasons: first because of its high Isp (>3000 sec) and second because Deep Space 1 (DS-1) is showing its capability as the main propulsion unit.

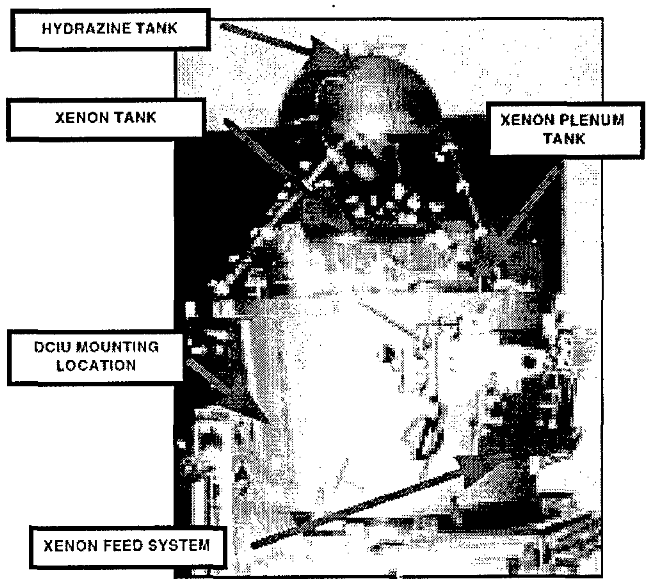


Figure 4.4-7. Propulsion Module Undergoing Integration at JPL

The challenge with electric propulsion was that very low thrust creates long orbit transfer times. Therefore, we sized the propulsion unit based on an acceptable time to transfer between the 5 planes. We chose the Scaled Down Ranger as the largest RS with electric propulsion. Based on 217 kg for the RMS and RS bus, the ion engine should be three times the size of DS-1, with a thrust of 0.3 Newtons. This corresponds to a complete tour of the constellation in 3 years (see spreadsheet 4). Since an ion engine requires a large amount of electrical power (2.4 kW for DS-1), we included the extra power system mass as part of the ion system. Also, since the ion engine has such low thrust, a small liquid propulsion unit would perform the final rendezvous burns. While either the ion or liquid system could perform the phasing maneuvers, for ease of calculations, we had the liquid propulsion system perform those maneuvers. An ion engine &

power system three times the size of DS-1 would have a mass of 320 kg. DS-1's ion engine and power masses were:

Solar Arrays: 27.7 kg ("DS 1 Solar Array Specifications", 1998)

PCU: 50.0 kg (assume .02 kg / Watt [Larson and Wertz, 1992: 319])

Ion engine: 41.3 kg ("Xenon Ion Propulsion System")

4.4.4.2 Spreadsheet #3: Phasing Within the GPS Orbit

4.4.4.2.1 Analysis Procedure

The two important characteristics for the propulsion systems are time and mass. Unfortunately, these two values are very interrelated, with propellant mass dependent on time needs and the time requirements dependent on the propellant mass. In addition, each alternative architecture has different time requirements. Thus, to make this a simple, yet accurate analysis we made it a two-step procedure.

In step one, we create a generic table of mass ratios based on a set of times. Thus time was the independent variable, and mass ratios the dependant variable (Appendix C-1). In step two, we pick a phasing time for the different architectures based on the generic mass ratios of step one (Appendix D-1). Then we calculate the real mass ratios for that specific architecture.

Each of the architectures has a unique mission, and so its mass ratios are different than step one (see below). In addition, alternatives have other variable dimensions like size of the robotic servicer and ORU total mass. Therefore, many iterations of this spreadsheet were used. Below is a description of spreadsheet 3 appendices, which are a sampling of the different alternatives.

Table 4.4-2. Diversity of Uses for Spreadsheet #3

Appendix	Servicer	ORU capacity	Plane	Used in Architecture
D-2	Low	50 kg	6	A, B
D-3	Med.	50 kg	3	B
D-4	Med.	150 kg	3	A, B
D-5	Med.	300 kg	6	A, D
D-6	High	85 kg (average)	3	D
D-7	High	300 kg	3	A, B

For the characteristics of the propulsion systems we used the same data resources as the canister's propulsion system from the logistics and transportation system. One important difference is the different structural ratio definition as defined in the beginning of this section.

4.4.4.2.2 Assumptions

While the GPS Satellite Vehicles (S/V's) are not spaced 90 degrees apart in their orbital plane, we used 90 degrees because that is their average spacing.

We used an elliptical phasing orbit. This required only two burns (one to get in the phasing orbit, one to get out). This phasing orbit was simpler to model. In reality, this system would probably use a circular-phasing orbit, because it would be more fuel-efficient. However, this requires four burns and was more complicated to analyze.

For more fuel efficiency, the RS intercepts S/V's behind it in the orbital plane. To intercept a target behind the RS, it would need a larger phasing orbit. A larger phasing orbit was more efficient than a smaller one because the period changed more quickly for a given Delta V. To confirm this, consider a LEO, GPS, and geostationary orbit. As shown in the LTS section, 75% of the required velocity change to get to geostationary was required just to get to GPS orbit. However, the change in orbital period from LEO to GPS was only 10.5 hrs (12 hrs. minus 1.5 hrs.). In comparison, the change in orbital period from GPS to geostationary orbit was 12 hrs (24 hrs. minus 12 hrs.).

4.4.4.3 Spreadsheet #4: Direct Plane Change Upgrader (Appendices E & F)

This spreadsheet calculates the propellant mass requirements for the two architectures that perform direct plane changes between GPS orbits. The first architecture uses ion electric propulsion (Architecture G) for these maneuvers. The second one uses solar thermal propulsion (Architecture H).

Ion propulsion has great specific impulse but very low thrust. Thus, we designed an ion propulsion system three times the size of DeepSpace 1's engine (see Section 4.4.4.1). The process of analyzing ion propulsion is described in Appendix E. The thrust levels of solar thermal make it a cross between high thrust chemical rockets and low thrust electric rockets. To analyzed its performance, we used a combination of impulsive and low thrust calculation techniques (see Appendix F).

4.4.4.4 Spreadsheet #5: Propellant Cost for On-orbit Depot (Appendix G)

4.4.4.4.1 Objectives

This spreadsheet is developed to calculate the RS propellant cost for one on-orbit depot (Architectures E & F). These architectures use chemical and solar thermal RS propulsion. In Section 4.5 it will be shown that these missions require a propellant re-supply mission due to the large quantities of propellant used. The output variable will be the size of the propellant re-supply mission.

4.4.4.4.2 Inputs

Like spreadsheet #4, this spreadsheet is linked to the orbital phasing maneuvers calculated in spreadsheet #3. Thus most of the input parameters for this analysis comes from input and output variables in spreadsheet #3. An independent variable that needs to be chosen is the structural mass ratio for the RS and re-supply tanks. Based on SMAD (p 660), we choose the

structural ratio to be 7% for the chemical propulsion tanks. Recall the chemical propulsion system uses nitrogen textroxide (N_2O_4) and monomethyl hydrazine (MMH), and the tanks can be compact and represent mature technology. However, solar thermal propulsion uses liquid hydrogen, which has a low density and is a cryogenic. Thus, the tank mass ratio will be 10%.

Like the dilemma in earlier alternatives, the mass of the propellant tank is a function of the required total impulse, which depends on the total mass. However, total mass is a function of the propellant tank. Thus, in the spreadsheet's input section, the spreadsheet's user needs to estimate the tank capacity and then compare it to the needed capacity and iterate until the two values match.

4.4.4.4.3 For Further Research

With the large amount of Delta V (12 km/s for 3 round trips from the 28 degree precessing orbit) it would be preferable to use a higher Isp propulsion system than the two systems listed here. Ion propulsion would be an excellent candidate, and we started this architecture with that propulsion system. However, with the low thrust, the maneuvers required for the RS to service a GPS plane and then to reinsert itself in the precessing orbit were quite complex. This complexity multiplied by the more elaborate methods of analyzing low thrust orbital maneuvers made this alternative beyond the scope of a top-level design.

4.4.4.5 Spreadsheet #6: Quick Lookup Tables for ORU Size Versus Canister Size

This spreadsheet is a multiplication table of ORU masses and number of S/V serviced. The middle portion of the spreadsheet is a reminder of number of S/V's serviced from spreadsheet #3. The bottom portion calculates average ORU mass and total mass for combined repair and upgrade missions. See Appendix H for spreadsheet #6.

4.4.4.6 Cost Modeling for the Different Propulsion Systems

4.4.4.6.1 Solar Thermal Propulsion Unit

We used the SOTV 190 kg mass for calculations of costs. This includes its propellant tank, which I did not include in an earlier section. The reason is each alternative has different tank sizes, but the cost of the tank is not a driving factor. With the RS costing method being fairly complex (Section 4.5) this assumption was beneficial. This mass can be used in the cost relation equations that were developed in the Logistics and Transportation System. By inserting 190 kg into spreadsheet #1 for the upper stage mass, we calculated a RDT&E cost of $22 million and a first unit cost of $4.7 million. To verify this number, we compared it with AFRL's Solar Orbit Transfer Vehicle (SOTV), which has a cost of $30 million for development and production (Dornheim, 1998:77). The SOTV cost includes a 285 kg bus subsystem that we have included in a different section from the RS cost.

4.4.4.6.2 Xenon Ion Propulsion

Our most in-depth costing tool, the NASA / Air Force Cost Model 1996 (see Cost Modeling, Section 4.4.9) has cost estimates for a variety of satellite components; however, it does not have any estimates for electric propulsion. However, the xenon ion system for Deep Space 1 (DS-1) exhibited many similarities to an electrical power distribution, regulation, and control (EPDRC) unit for a xenon ion propulsion system. The EPDRC cost relationships were a useful way to calculate the cost of the ion propulsion systems. Since ion propulsion was a new technology, we used the higher planetary cost relations instead of the lower earth orbiting satellite cost relations. The mass of an ion propulsion system was three times the size of DS-1, which was 273 kg or 601 lbs. See Appendix E-1 for the electric propulsion spreadsheet calculations.

Since electric propulsion used a tremendous amount of electrical power, it was logical to calculate the cost for the additional size in the solar arrays in this section. Considering the robotic servicer is orbiting the earth, we used the traditional earth orbiting cost relations in NAFCOM. The mass of solar arrays was 83 kg or 183 lbs.

The NAFCOM calculated a cost of $27.3 million for DDT&E and $7.9 million for production of the ion propulsion and additional power system (Spreadsheet #23). This corresponds to $38 M for total cost of DS-1's ion engine (Dornheim, 1998:108). While DS-1's ion engine was smaller, it was the first of its kind and thus the difference is reasonable.

4.4.5 Robotic Manipulating & Bus System Analysis Procedure

4.4.5.1 Definitions

Since space robotic servicing systems are not a common discipline, we have included the following definitions to make the analysis clear.

DEXTEROUS ARMS: The robotic arm(s) that manipulate the GPS satellite in the servicing of selected components.

END EFFECTORS: The "hands & tools" of the dexterous arms that will enable the RMS to open access doors, manipulate thermal blankets, disconnect electrical connectors, unbolt ORUs and handle ORUs. Mostly likely one (or both) dexterous arm(s) will have to be able to use multiple end effectors for the different tasks.

GRAPPLE ARM: A manipulator that attaches to the GPS S/V and repositions the RS to the work site. This arm enables the RS to work on parts of the GPS satellite somewhat distant from its attachment point. For University of Maryland's Ranger Program this is a 7-Degree of Freedom manipulator.

ORBITIAL REPLACMENT UNIT (ORU): A component or black box on the GPS satellite vehicle (S/V) that will be removed and replaced. Since most GPS components are packaged in electrical boxes, they can also be called black boxes; however, they could also represent non-box like components like reaction wheels.

POSITIONING ARM: A robotic arm that moves the RMS to the work-site once the RS and GPS S/V are docked. A positioning arm is different from a grapple arm in that it only moves the RMS to the work site; whereas the grapple arm moves the entire RS to the work site. An example of the difference is the configuration of Ranger versus the configuration of the Flight Telerobotic Servicer.

RS BUS: The subsystems on the RS that provide power, ground communication, navigation, attitude control, close proximity maneuvering, and the ORU Storage System (OSS).

ROBOTIC MANIPULATING SYSTEM (RMS): The RS's payload, which includes the dexterous arms, end efforts, robotic vision system (RVS), grapple arm or positioning arm (if needed), and the task interactive computer (TIC).

ROBOTIC SERVICER (RS) – The entire spacecraft that does servicing, including the RMS, the docking unit, the bus, and propulsion unit. The RS is a sub-component of the overall Robotic Servicing System (RSS).

ROBOTIC VISION SYSTEM (RVS): The video camera(s), the camera's support arm(s), lighting, and other sensors needed for the RMS to perform its duties

RS TRANSPORT VEHICLE (RTV): For the free-flying servicer configuration, this would be the "mother" vehicle for the SMS. This provides the power generation, data management, orbital maneuvering propulsion system, ORU pallet, and the bulk of the data management and communication.

SERVICING MICRO-SATELLITE (SMS): Composed of the RMS and support systems, this satellite would detach from the RS and service the GPS S/V. The benefit of this configuration is this could be much smaller than the entire RS and thus more easy for docking and servicing. (Not used in every alternative)

TASK INTERACTIVE COMPUTER (TIC): The TIC is the processor that control the manipulators. Whether automated or teleoperated, operational robots have a feedback loop that is not feedback to the user. The reciprocal of the TIC is the ground-based Human-Interactive Computer (HIC). The HIC sends the appropriate commands from the human operators to the RS.

4.4.5.2 Analysis Procedure

4.4.5.2.1 Scope

The RMS is the payload for the Robotic Servicer (RS) and is the key interface between the GPS S/V and the RSS. The RMS system design interfaces directly with the Aerospace Corporation (ASC) study on GPS S/V serviceability. By defining a few different RMS concepts as a basis for the RSS and GPS studies, the two studies will compliment each other. My plan was to define a few alternative concepts for servicing. Then with ASC and the leading university in space robotics research, we outlined a few general specifications for each alternative RMS.

This could be an entire thesis unto itself; however, since our thesis objective is to analyze the overall system, this step is to define what is or projected to be available in the industry and gain some broad, general parameters by which to perform the rest of our research.

4.4.5.2.2 Approach

Current robotic servicing programs are demonstrating capabilities; thus, they are focused on what technology can do, and not on what tasks we need technology to do. There is not a set

of equations that describe the system level characteristics of a robotic servicer. Textbooks like Space Mission Analysis and Design (SMAD) provide characteristic equations for other space systems. Without these equations, the characteristics of robotic servicers are more difficult to determine. This thesis will use characteristics based on current or previous robotic servicers. These characteristics will then be modified to account for the unique requirements of servicing GPS S/V's. These unique requirements include orbit and design life, both of which directly impact the servicer's bus subsystem. SMAD then provides the techniques to modify previous servicer bus subsystems to meet these new requirements.

Even with this methodology not all characteristics can be determined quickly. Therefore, this study will analyze the important characteristic variables. Which characteristics are important depend on their impact on the cost and performance of the overall servicing system, as well as the what variation these characteristics can exhibit.

In addition, in order that our study can be utilized with Aerospace's study, we define key interfaces between the robotic servicer and the GPS S/V. We defined these interfaces as robot – satellite interface variables (RSIV) (Section 4.3.5.4).

4.4.5.3 Characteristic Variables

4.4.5.3.1 Mass of RMS

The importance of mass of the RMS will be affected by whether or not the RS maneuvers between planes. If there is a RS in every plane, the mass will still be somewhat important because of the high cost to transport the RS to the GPS orbits. The mass varies due to RS-RMS interface, the reposition system, support requirements of the RMS, and size of ORUs handled.

4.4.5.3.2 Mission Costs

Compared to the magnitude of cost for the other RSS components, mission costs should not be a dominant variable. Much of the time the RS will be in transit between GPS satellites and will need minimum oversight. This should minimize the 24 hr. manning requirement. The trade study of ground operations manning versus automation of the servicer should have little impact on overall system costs. In a worst case alternative, the RS would be highly complex and rely on telerobotics. This is analogous with the Ranger Telerobotic Servicer. In their TFX program their baseline was 2 teams of 6-8 people per team ("Ranger Telerobotic Flight Experiment IDR," 1996: 55,56). For a 50% mix of government and contract personal this would correspond to $940,000 / year (Larson and Wertz, 1992: 730). If a highly automated servicer with some type of supervisory control could reduce this manning by 50%, the savings would be approximately $7.5 M over 15 years.

4.4.5.3.3 Mission Timeline

Compared to timelines involved in other RSS elements, the servicing time requirements should have a low overall impact. For example, for the RS to maneuver between planes takes 30-60 days. An average servicing mission was 1-5 days (3 days average). These numbers came from the two Hubble servicing missions, which required 35.5 hrs of EVA on the first mission and 33 hrs on the second mission (Waltz, 1993: 286,288). Also, Ranger Telerobotic Shuttle Experiment missions were allocated 36 hrs for operations.

4.4.5.3.4 RS Cost

Cost is an important variable for any program. The main method that we used to determine these cost was the NASA / Air Force Cost Model 96 (NAFCOM). Science

Applications International Corp. developed this cost estimation program for the government using a large database of previous programs.

4.4.5.3.5 Design Life

While the hardware of the RMS should be fairly resilient, the RMS will need a fair amount of processing power for the TIC, which is sensitive to the space environment. Also, this variable will depend on the RS bus and propulsion design, which I will design in a different section.

4.4.5.3.6 Percent of GPS Serviced

Percent of GPS serviced is an important characteristic. In addition, it will be highly dependent on the design of the GPS serviceable satellite, and the RS. This variable will probably be the biggest delineator between the different RMS alternatives. Finally, the percentage will be highly correlated to the employment strategy chosen. See Section 4.4.5.4.1 for further elaboration of the servicing configuration of the GPS S/V.

4.4.5.3.7 Percent Success Rate

Two issues drive why percent success rate will not be studied. First, the range between the values will probably be low (95-99%). Second, and more importantly, the level of design for ~~a~~ this systems type analysis would not be detailed enough to ~~delineate~~ discriminate any difference in success rate.

4.4.5.3.8 Summary Table

The following table outlines the range of values for the above variables. As mentioned before, it would be beneficial not to model every variable, so the last column indicates which variables were used and if they were an input or output.

Table 4.4-3. Characteristic Variables

	Decisive	Varies	Chosen Variables
Mass of RMS	High (plane change RS) Med. (no plane change RS)	High	Output #1
Mission cost (ground crew, Comm. Support)	Low	Med.	
Time for Service Mission	Low	Med.	
Cost - Development & Production	Med.	Med.	Output #2
RS Design Life	Med.	High	Input #1
% of a GPS S/V serviced (capability of servicer)	High	High	Input #2
% success rate	Med.	Low	

4.4.5.4 Satellite – Robotic Interface Variables (SRIV)

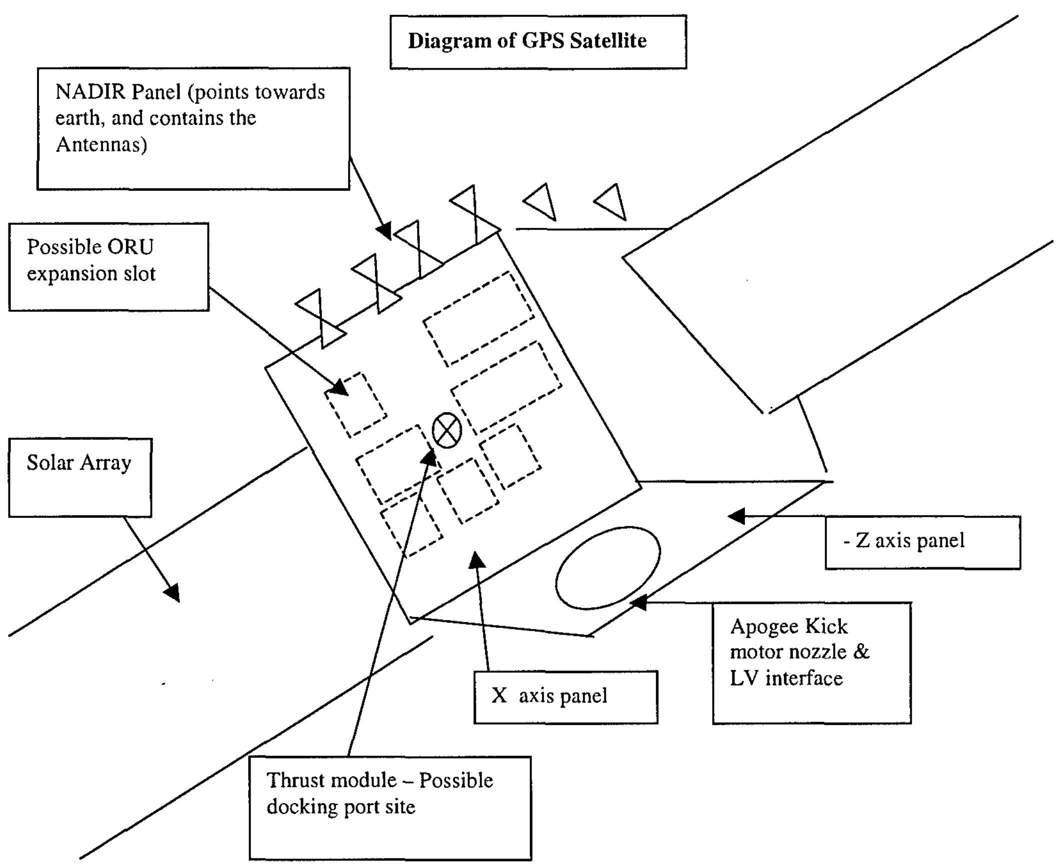


Figure 4.4-8. Diagram of GPS Satellite

4.4.5.4.1 Configuration of Serviceable Components

Configuration of serviceable components would be a primary determinant for percentage of the GPS S/V serviced, complexity of the RS, and many other variables. The companion Aerospace Corporation study defined this variable. The study assessed impacts on making the GPS S/V serviceable. The "no change" option would require the RS to be the most dexterous and change-out components on GPS the way humans can on the ground. The "exterior mount" would add shielding to the ORU. It would also require serviceable components to be located on the exterior. This would limit the number of serviceable components. The "internal mount on replaceable panels" option would require the + / - X panels of the GPS S/V to be like a checkerboard with each removable panel having an ORU on the inside. The six sides of the GPS S/V are described by a body mounted frame with X, Y, Z axes. Thus the + / - X panels are perpendicular to the X axis of the satellite's body frame. The "drawer" option was similar but would reduce the precision requirements for the RMS. The access door option would locate the ORUs either inside on the door or in an easily accessible location. This method might be the easiest attaching mechanism for the ORU, but it would require a fairly dexterous RMS.

4.4.5.4.2 Docking Location

This variable is dependent on ORU accessibility and the necessity to dock with spinning S/V. If the RS docks on the –Z-axis then a RMS positioning system is needed. If RS docks on the side panel a positioning system could or could not be included. If the RS only is needed to dock with a 3-axis stabilized GPS S/V then either location is permissible. If the S/V is spinning then only a –Z-axis docking location is acceptable.

4.4.5.4.3 Attitude Control for the Combined RS & GPS S/V

This variable determines the attitude control system when the RS and S/V are docked. Mostly likely only control through reaction wheels or momentum wheels will be acceptable. One reason is possible contamination of the internal GPS S/V from thruster exhaust. Another reason is the GPS S/V will not be able to use its propulsion system because it is located on the panels with the access doors.

If the GPS S/V controls attitude then the docked RS mass will have to be under a certain limit. This limit will be determined by the maximum mass the GPS S/V attitude control system can handle. The most likely way to get the RS mass under a certain limit is to have it configured with a detachable Servicing Micro-Satellite (SMS).

4.4.5.4.4 Break-out Box Capability

Break-out Box capability means the servicer could detach a coaxial cable from a GPS component and route the signal through itself back to its component. Then the servicer could downlink the signals to an engineering team on the ground. This was a common practice at the factory and Cape Canaveral for trouble-shooting anomalies on satellites.

Aerospace's study was not complete in time to incorporate their results. Their study may show break-out-box capability to be an important variable. However, we left it as an area for further study.

4.4.5.4.5 Solar Array or Antenna Replacement

In both examples, the RS would need some type of reach capability from the docking location to the area receiving service. The reason is that the delicate nature of solar arrays and antenna requires precise maneuvering around these components. In effect, docking or free

flying servicers around these components on the GPS S/V would bring inappropriate risk to damaging these components.

4.4.5.4.6 Summary Table

Below is a summary table of the Satellite Robotic Interface Variables.

Table 4.4-4. Satellite Robotic Interface Variables (SRIV)

<table>
<tr><th>Interface Variables</th><th colspan="8">Options</th></tr>
<tr><td>Configuration of serviceable components</td><td colspan="2">No change</td><td>Exterior Mount</td><td colspan="2">Internal mount on replaceable panels</td><td colspan="2">Drawers</td><td>Access doors</td></tr>
<tr><td>Docking location</td><td colspan="3">Side panel fixture (+ / - X panel)</td><td colspan="5">- Z axis of GPS S/V (e.g. S/V to booster interface ring)</td></tr>
<tr><td>Combined attitude control</td><td colspan="3">GPS (using only reaction wheel control)</td><td colspan="5">RS controlled</td></tr>
<tr><td>Solar array or antenna replacement</td><td colspan="3">None</td><td colspan="3">Antenna only</td><td colspan="2">Antenna & solar array</td></tr>
<tr><td>Break-out box capability</td><td>No</td><td colspan="3">Yes, for a few connectors</td><td colspan="2">Yes, for many connectors</td><td colspan="2">← For further research</td></tr>
</table>

4.4.5.5 RS Configuration Options

4.4.5.5.1 RS-RMS Interface

The RS-RMS interface variable determined if the entire RS docked with the GPS S/V or only with a servicing mini-satellite. The main advantage of having a SMS was that hopefully the GPS S/V could control attitude. This variable also depended on whether or not the RS had to maneuver between GPS orbital planes.

4.4.5.5.2 RMS Positioning System

This system positions the RMS to the appropriate work site on the GPS S/V. If the ORUs were next to the docking port, a positioning system would not be necessary. However, if the RMS needs to open access doors or docks with the –Z axis panel of the S/V, a positioning system is necessary. An example of a grapple arm that maneuvers the entire RS is University of

Maryland's Ranger Telerobotic Servicer. Ranger has a 7-degree of freedom (DOF), 84" grapple arm (Ranger TSX, 1997:3). An example of a positioning arm is NASA's Flight Telerobotic Servicer (FTS). In one example of an OMV-FTS design, TRW based-lined a 5 DOF, 6 meter positioning arm (Waltz, 1993: 211). Hand over hand maneuvering are represented in the concepts proposed by the Air Force Research Laboratory's Modular On-orbit Servicing (MOS) Integrated Product Team.

4.4.5.5.3 Docking Mechanism

The two categories of docking methods are the traditional probe in docking port used for 30 years in manned space-flight, and the more controlled robotic grasping method used by the Space Shuttle's robotic arm with various user satellites. For our application the grasping method could be done by the grapple arm (if available) or one of the dexterous arms. If done by the dexterous arms then the RS would need a second attachment system so as to free up the dexterous arm for its servicing mission.

4.4.5.5.4 RS Configuration Variables Summary

Below is a summary of the RS Configuration Variables.

Table 4.4-5. RS Configuration Variables

<table>
<tr><td>RS-RMS interface (dependant on SRIV variable #3)</td><td colspan="2">Attached (the RS will control attitude of the combined RS GPS S/V system)</td><td colspan="2">RMS detaches from RS to dock with the GPS S/V (the GPS S/V will control attitude)</td></tr>
<tr><td>RMS positioning system (from docking to servicing location) (dependant on SRIV #1 and #2)</td><td>None</td><td>Grapple arm moves entire RS</td><td>Positioning arm moves only RMS</td><td>Hand over hand maneuvering</td></tr>
<tr><td>Docking mechanism (dependant on SRIV # 2)</td><td colspan="2">Dexterous arm</td><td>Grapple arm</td><td>Probe</td></tr>
</table>

4.4.5.6 Process for Synthesizin g into Alternative Categories

4.4.5.6.1 Objective

Recall that the objective for the thesis is to analyze different top-level architectures of on-orbit servicing. It was neither necessary nor beneficial to design all the different alternative servicing methods available. Instead, this section characterizes benefits and costs of different servicing methods. Thus, this study will outline three typical RMS's to represent three performance categories of RS. The three categories are a high, medium, and low capability robotic servicers.

4.4.5.6.2 Environmental Scan

The most positive way of getting verifiable characteristics of alternatives without designing them is to find past or current robotic space servicers that could be used in our application. While Chapter 2 focus on the history of on-orbit servicing, this looks at specific configurations we can use for this study. Four programs similar to our application are the following: the Flight Telerobotic Servicer, the Special Purpose Dexterous Manipulator, Robonaut, and Ranger.

Orbital Maneuvering Vehicle - Flight Telerobotic Servicer (OMV-FTS). This was a very ambitious on-orbit servicing program about 7 –10 years ago. This program contributed significantly to the knowledge of on-orbit servicing (Waltz, 1993: 202-213). However, for two reasons it seemed inappropriate to use OMV-FTS for a baseline for this study. First, it was a Space Shuttle based system and with GPS being located at semi-synchronous orbit, the OMV did not have performance to rendezvous with those satellites. Second, robotics is a relatively new field and base-lining a servicing system in the future on technology 10 years in the past seemed unwise.

A current and important robotic servicing program is the International Space Station's Special Purpose Dexterous Manipulator (SPDM) by Canada. The SPDM is to perform servicing on the space station to minimize the EVA time needed for maintenance. However, SPDM will always be attached to the ISS by the Mobile Servicing System (MSS), and receive all it's power, command signals through the MSS (Asker, 1997:71,73). Thus, the SPDM is not designed with the docking, maneuvering, power generation, communication and other requirements needed for satellite servicing. Nevertheless, SPDM is contributing significantly to on-orbit servicing and this study uses one of its end effectors called the ORU Tool Change-out Mechanism (OTCM) (Sullivan, 1998).

A follow-on space station servicer to SPDM is a NASA program called Robonaut. Its objective is to develop a servicer that could perform short-sleeve human servicing tasks through advance robotic technology (Parish, 1998). This would greatly reduce the cost of making space vehicles serviceable. However, with the challenges in technology development, Robonaut is not a good baseline to understand the system level costs and benefits for on-orbit servicing in the near future.

With the cancellation of the OMV-FTS, NASA's Space Telerobotics Program focused its resources on more experimental programs. University of Maryland's Ranger Telerobotics Experiment is the most ambitious of these programs. Ranger was originally to be a free-flying experiment called the Telerobotics Flight Experiment (TFX). The TFX was to be launched on Lockheed Martin's Launch Vehicle (now called Athena). However with a $ 20+ M launch cost, NASA balked at the cost. Therefore NASA redirected Ranger to be a Shuttle experiment (called TSX), since Shuttle costs are internal to the agency. TSX is slated for a Shuttle mission in the fall of 2000 (Parish, 1998). One advantage of placing Ranger on the Shuttle rather than a

expendable launch vehicle is that Ranger will have to demonstrate the fail-safe modes of maneuvering manipulators and other objects in close proximity of a manned space vehicle. Since the original TFX had gone through preliminary design, it is perhaps the best baseline of a robotic on-orbit servicer. For this reason the Ranger TFX servicer will be one of the three baselines described below.

4.4.5.7 Alternative Robotic Servicers

This section describes the overall concepts of the three types of servicers. It describes the interface of the 3 categories of RMS with the GPS S/V. However, the analysis of the 3 RS for the 3 RMS's will be analyzed in the following section (Sections 4.4.6 – 4.4.8).

4.4.5.7.1 An Operational Ranger (High Performing Servicer)

Ranger's performance requirement is to perform the dexterity, strength, and reach envelope of a space suited astronaut. One difference from an EVA astronaut is that Ranger would use a collection of mechanical tools as end effectors instead of the highly dexterous five-fingered hand.

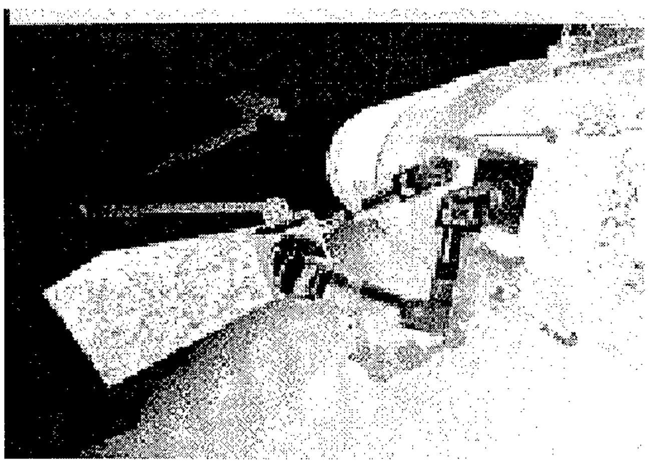


Figure 4.4-9. Ranger in Action

With a grapple arm, Ranger could dock with a docking mechanism on the +/- X panel of GPS S/V or the launch vehicle interface ring on the –Z axis of the GPS S/V. For the above reasons,

the Operational Ranger will be characterized as being able to change-out ORUs in all the serviceable component configurations except the "no change" option (see Appendix Y).

4.4.5.7.2 The Scaled Down Ranger (Medium Performing Servicer)

The configuration of servicing options that Aerospace Corp. has generated has provided the option of scaling down Ranger while being able to change-out ORUs in many of the serviceable component configurations. For this servicer the drawer or replaceable panel design would be the configurations of the ORUs. These two methods reflect the type of ORU than can be changed-out by SPDM's ORU Tool Change-out Mechanism (OTCM). The OTCM has both the ability to grasp ORUs and manipulate the ORU fasteners all on one end effector. Thus, for simple tasks like ORU change-out in the drawer or panel configurations only one manipulator arm is needed. Also the Aerospace study suggests having the docking port in the center of the +/- X panel of GPS. Thus, we can eliminate the requirement for the grapple arm by placing the drawers or panels for the ORUs directly around the docking port. By reducing the complexity of the task and reducing the number of robotic arms in half, we should reduce the communication, data management, and power requirements on the RS bus by at least half.

4.4.5.7.3 The Free Flyer Servicer (Low Performing Servicer)

One concept that has been proposed by the AFRL's Modular On-orbit Servicing (MOS) Integrated Product Team (IPT) is a free flying servicer (Madison, 1998: 4.2.4). With the maturing of docking techniques and the ability to grasp and fasten ORUs with one end effector it should be possible for a small free-flyer to dock the ORU directly into its intended slot. The advantage is we reduce the robotic servicer's manipulation requirements to only docking. The disadvantage is that the ORU electrical and mechanical connections to GPS will also need to be a docking port. One solution is to have the ORU and accompanying slot or drawer on GPS to be

designed such that the RS can "soft" dock the ORU on GPS. Once in a "soft" docked configuration, the RS would actuate the ORU fasteners to provide a final "hard" dock configuration. The electrical connectors would be attached during the actuation of the ORU fasteners. Thus, the alignment requirements for these connectors would be meet by the design of guide rails of ORU drawer / slot and not be required of the "flying" ability of the RS.

The overall configuration of the RS is to have the Free Flyer (Servicing Micro-satellite) be maneuvered to different GPS S/V's by a RS transport vehicle (RTV). The SMS would detach, transport the ORU to the GPS S/V and insert it into the appropriate slot. The SMS would consist of a small maneuvering propulsion unit, camera, battery, processor, and OTCM to perform its mission. The RTV would provide the orbital propulsion unit, the bulk of communication system, the power generation from solar arrays, a camera to provide situational awareness to ground crew, and an ORU pallet.

There are at least two configurations for the ORU slot onboard the GPS. First is a design that places the slot directly on the external surface of the GPS S/V. This would require the mechanical and electrical connections to be hardened for the space environment. The second option is give the slot a cover panel, for which the SMS would have to dock and detach before inserting the new ORU. The timeline described below will use the second method.

One large advantage is the RS attitude control system would not have to be designed to control the combined GPS Satellite Vehicle / RS combination. One large disadvantage is there would be large risk in flying the SMS near the sensor and antenna panel (nadir panel). Therefore the ORU slots for the free flying servicer would need to be away from the nadir panel. Thus for change-out or addition of new sensors or antennas, this free-flying servicer would not be an option.

4.4.5.8 A Servicing Mission Order of Events for the Operational Ranger and Scaled-Down Ranger RS

To give a better understanding of the above concepts we will describe a notional servicing mission. The robotic servicer will also pick up ORUs from the canisters. An ORU pick up would require docking, transferring of ORUs and undocking; therefore, it is a simpler version of the procedure described below.

1. The RS docks with the GPS satellite, locates the RMS to the serviced area
2. Opens access doors (if needed)
3. Swaps out boxes
4. Closes door (if needed)
5. Detaches from GPS
6. Waits in close proximity until ground control powers up and verifies the GPS subsystem that was serviced
7. Either maneuvers to next task, or re-docks with GPS S/V for ground directed trouble-shooting.

4.4.5.9 A Servicing Mission Order of Events for the Free-flying RS

While the above two configurations have the same type of servicing scheme the Free-flying RS will have a different methodology.

1. The RTV would rendezvous to approximately 50 –100 meters away from the GPS S/V
2. The SMS would detach, dock to the cover panel on the ORU slot, detach it and return to the RTV. It would place the cover panel on the RTV or with a very small maneuver place it in a different orbit from the GPS S/V.
3. The SMS would then dock and disconnect the ORU from the RTV.

4. The SMS would maneuver to the GPS S/V and insert the ORU into its slot with a "soft" dock configuration.
5. The SMS would actuate the ORU fasteners to fully connect the ORU to the GPS S/V.
6. The SMS would detach and re-dock with the RTV
7. The RS waits in close proximity until ground control powers up and verifies the GPS subsystem that was serviced
8. The RS either maneuvers to next task, or re-docks with GPS S/V for ground directed trouble-shooting.

4.4.6 Operational Ranger

The Operational Ranger servicer alternative used University of Maryland's Space System Laboratory (SSL) Ranger Telerobotic Flight Experiment (TFX) as the baseline design. The data used in our study was taken from the Ranger TFX's Integrated Design Review (IDR) #2 presented on April 3-5, 1996. TFX is a good example of a high performance robotic servicer. Since TFX was designed to be a LEO 60-day experiment, the goal of this analysis is to determine the different characteristics of an operational Ranger.

Joe Parish, program manager for Ranger, and Gardell Gefke, deputy program manager, outlined two important differences between a university experiment and a commercial operation (Parish, 1998; Gefke, 1998). First, commercial program costs could be up to 100% above the costs of a university project. Later, this study (Section 4.4.9) found results similar to their prediction. Second; an operational Ranger would probably weigh up to 50% less than Ranger TFX would. The reason is Ranger TFX was based-lined to be launched on the Lockheed Launch Vehicle which had much more lift capability than Ranger needed. Thus the flight Ranger was designed the same as their underwater prototype version. An example of this design philosophy

is that the main structure was built out of ½ inch aluminum plate. This is much heavier than the honeycomb structures of most satellites. For this reason, our study started with an operational Ranger with a 40% reduction in mass compared to the TFX Ranger.

However, the reduced Ranger of this study will need extra mass to account for different orbit and design life issues. With design lives spanning 120 days to 15 years it was important to account for how these differences changed the characteristics of each subsystem. For most subsystems a general percentage increase was included to account for radiation hardening and redundancy.

Radiation hardening is important to consider because GPS's orbit is directly in the Van Allen Belts (Larson and Wertz, 1992: 200). Thus an operational Ranger will need more electromagnetic hardening that the Ranger TFX. This increase was calculated by a "Satellite Mass Increase to Hardness" table found in Space Mission Analysis and Design (221). However, even for a heavy radiation hardened robotic servicer (hardness level of $2*10^{-1}$ cal/cm^2), this only corresponds to a mass increase of 3.5%.

Redundancy is probably the most common way of ensuring high reliability for longer design lives (Larson and Wertz, 1992: 711). Component redundancy requirements can be determined using reliability analysis. However, reliability analysis is beyond the scope of this study. To capture the relationship between increase design life and redundancy, a mass percentage increase was included for each design life option.

The three subsystems that cannot be characterized by the general percentage increases are the propulsion, electrical power, and mechanical subsystems. Obviously, propellant needs would be based on total required changes in velocity (Delta V) and not on design life. Therefore, this requirement is calculated in the RS propulsion section of the thesis. The mechanical subsystem

does not vary much according to design life because most of its requirements are based on the launch environment. A small percentage increase was added to account for any quality issues (such as thermal stressing) associated with long-term missions (see Spreadsheets 7 – 9). Another subsystem subject to considerable variation is the power system.

We designed the electrical power system according to power requirements and design life. The Operational Ranger was designed with the same end of life (EOL) power requirement as the TFX Ranger (675 watts ["Ranger Telerobotic Flight Experiment IDR," 1996: 196]). This should be adequate since TFX's orbit average power requirement was 346 Watts. Like the TFX design, we chose solar arrays to be the power generation system. To compare three types of solar arrays with different design lives, we performed the design process in Space Mission Analysis and Design (p. 395) using Microsoft EXCEL (Spreadsheet #10). The three solar array types are silicon, gallium arsenide, and indium phosphide. We was able to estimate their power output / meter^2 by multiplying their efficiency value (Larson and Wertz, 1992: 397) by the incident solar radiation (1358 W / m^2). Next, we found the beginning of life (BOL) power capability by taking out the inherent degradation, and any incident angle losses. Within the design life table, we found the EOL power capability by multiplying the BOL capability by the total degradation of the arrays.

$$BOL*(1\text{-}degradation\,/\,year\,)^{satellite\ life} \qquad \text{Equation 14}$$

For silicon and gallium arsenide, SMAD had total degradation / year values and the degradation due to radiation. Their values were based on the Low Earth Orbit (LEO) environment. Since GPS orbit has a higher radiation environment, we increased the degradation due to radiation by 50%. For indium phosphide we calculated the degradation / year value from

the published fact that the total degradation in 89 years is 15% (Larson and Wertz, 1992: 397). We still increased this value by 50% to reflect the higher radiation dosage of a GPS orbit.

The required solar array area is found by dividing the power requirement by the value for the EOL power capability per unit area. This can be converted to mass by multiplying it by the specific performance of the planar array. The Firesat Satellite example in SMAD had 3.66 kg / m^2 for its solar arrays. We used 4.00 kg / m^2 to be conservative. Spreadsheet #10 is illustrated below.

Spreadsheet #10: Design of Robotic Servicer's Solar Arrays

(process taken from SMAD p395)

Incident angle=	5 degrees
Inherent deg =	0.82

Assumptions			
Solar cells: Silicon	eff =	0.14	
	Power output =	190	W/m^2
	Determine BOL power capability =	155	W/m^2
	Degradation / year =	0.0500	
Solar cells: Galium Arsenic	eff =	0.18	
	Power output =	244	W/m^2
	Determine BOL power capability =	200	W/m^2
	Degradation / year =	0.0350	
Solar cells: Indium Phosph	eff =	0.19	
	Power output =	258	W/m^2
	Determine BOL power capability =	211	W/m^2
	Degradation / year =	0.0027	

Requirement	
RMS & bus (W) =	675
ion drive =	0
total =	675

*note: for degragation/year I took SMAD's LEO numbers (p 400) and increased radiation degragation by 50%.

Design life	0.5	2	10	15	Ranger TFX example
EOL/Area (Silicon) =	151	140	93	72	
EOL/Area (GaAr) =	196	186	140	117	
EOL/Area (Ind. Ph.) =	210	210	205	202	
Area (Silicon) =	4.5	4.8	7.3	9.4	5.95
Area (GaAr) =	3.4	3.6	4.8	5.8	
Area (Ind. Ph.) =	3.2	3.2	3.3	3.3	
Specific performance of planar array		4 kg/m^s			
Mass (Silicon) [kg] =	18	19	29	38	24
Mass (GaAr) [kg] =	14	15	19	23	
Mass (Ind. Ph.) [kg] =	13	13	13	13	
Percentage Increase (GaAr) =		0.05	0.40	0.68	

Figure 4.4-10. Solar Array Calculations

The power storage requirement (i.e. batteries) for the Operational Ranger will differ dramatically from TFX because of orbital differences. Since TFX was in a 90 minute LEO orbit, it had to perform robotic servicing in or out of sunlight. For the GPS orbit, a satellite is in maximum eclipse for only one hour in a 12-hour period. Therefore the Operational Ranger can perform its servicing only in sunlight. Thus, the power requirement during eclipse is only from the RS bus. This dramatically reduces the power storage requirement since the robotic manipulators require 83% of the nominal power ("Ranger Telerobotic Flight Experiment IDR," 1996: 197). Therefore, we reduced the 3.2 KW*hr power storage requirement of TFX Ranger to 544 W*hr for the Operational Ranger. However, GPS eclipse time is 50% greater than the LEO eclipse time (60 versus 38 minutes [Larson and Wertz, 1992: back cover]). Hence, the final requirement for the Operational Ranger was 860 W*hr.

We used two types of batteries based on robotic servicer design life (see spreadsheets 7 – 9). For the short design life (120 days), we used the same batteries as the TFX Ranger, Silver Zinc. These batteries have short lives, but high specific energy density (60 – 130 W*hr/kg [Larson and Wertz, 1992: 402]). For longer missions of 2 to 15 years, we used nickel hydrogen with individual pressure vessel design. They have lower specific energy density (25 – 40 W*hr/kg) but are used in long term missions. Unlike the 2-year mission, which can allow 100% depth of discharge, the 15-year mission batteries should only be used at 50% depth of discharge (Larson and Wertz, 1992: 318). The reason for a lower depth of discharge is in 15 years, the batteries will have over 10,000 cycles. This doubles the size and mass of the 15-year batteries verses the 2-year batteries.

The Electrical Power System (EPS) contains many components in addition to the solar arrays. These components include the power conditioning, distribution, and regulating system.

These components have similar characteristics to the generic subsystems and thus will use the general percentage increase. The method to calculate the TFX's masses for those components is to use the EPS mass of TFX and subtract the solar array and battery mass for TFX. The IDR data for TFX did not have those components' masses, so they were calculated using the methodologies outline above. For silicon solar arrays this corresponds to a mass of 51 kg (See SS #10). The battery's mass was calculated by the values for Silver Zinc batteries

*3.2 KW*hr / 90 W*hr/kg = 35.5 kg or 78.4 lbs* Equation 15

Thus, the rest of the EPS subsystem had a mass of 121 lbs (250 – 50 - 78.4 = 121).

The breakdown of mass for each subsystem for the TFX ("Ranger Telerobotic Flight Experiment IDR," 1996: 31) was used for a baseline for spreadsheet #7. Components with a typical relationship between mass and design life were multiplied by the total general percentage increase. This represented the top third subtotal. The subsystems that required special calculations represented the middle subtotal. The payload was the final subtotal and was also multiplied by the general percentage increase factor. Spreadsheet #7 is in Figure 4.4-14 below.

Spreadsheet #7: Operational Ranger (High performance)

Percent mass reduction to make Ranger operational = *40* %

	Structural % increase	% rad.hard	% redundancy.	% total	(hardness figures come from SMAD, p. 221)
120 days	*0*	*1*	*5*	6	hardness level: 10^-3 cal/cm^2
2yrs	*2*	*1.3*	*10*	11.3	hardness level: 10^-2 cal/cm^2
10 yrs	*4*	*2.3*	*15*	17.3	hardness level: 10^-1 cal/cm^2
15 yrs.	*6*	*3.5*	*30*	33.5	hardness level: 2*10^-1 cal/cm^2

Electrical Power Supply Requirements			
Power = 675 W	120 day (AgZn)	2 yr (NiH-100% dis.)	15 yr (NiH-50% dis.)
Battery discharge energy = *860* W*hr	10 kg	19 kg	38 kg

Summary of Subsystem Masses for Operational Ranger

	pounds		mass reduction	kgs.	120 day design life	120 day (lbs.)	2 yr. design life	lbs.	15 yr. design life	lbs.
Spacecraft Bus		852	511	232	246	542	258	569	310	682
Data Management System	14		8	4	4	9	4	9	5	11
Thermal	32		19	9	9	20	10	21	12	26
Communication System	30		18	8	9	19	9	20	11	24
ADACs & RCS	272		163	74	78	173	82	182	99	218
EPS (-solar arrays & batt.)	*121*		73	33	35	77	37	81	44	97
Sub Total =		469	281	128	135	298	142	313	170	376
Battery	*78*				10	21	19	42	38	84
Solar arrays (silicon)	*51*		(Gal. Ars->)		14	31	15	33	23	51
Structures & Mechanism	251		151	68	68	151	70	154	72	160
Sub Total =		302			92	203	104	229	134	295
Payload		524	314	143	151	333	159	350	190	420
Dexterous Arms	232		139	105						
Grapple Arm	130		78	59						
Video Arm & Cameras	106		64	48						
End Effectors	56		34	25						
Total mass (in kgs.)		1376	826	349	378		405		494	

Figure 4.4-11. Summary of Subsystem Masses for Operational Ranger

4.4.7 Scaled Downed Ranger

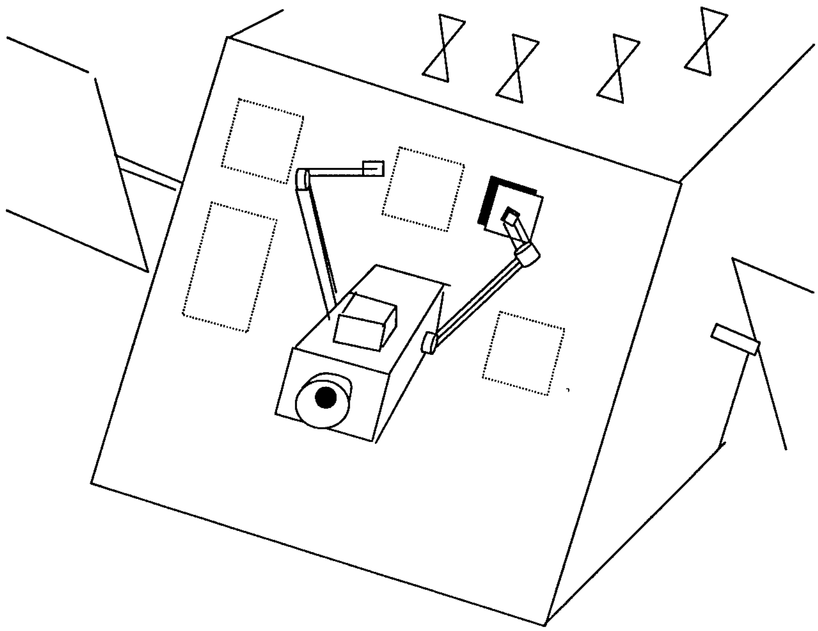


Figure 4.4-12. Diagram of a Scaled Down Ranger concept

To represent a medium performance class of robotic servicers, we reduced the complexity, capability, and size of the Operational Ranger. The essence of this idea is that a combination of the OTCM and Aerospace's replaceable panel configurations (Section 4.4.5.8) could permit a scaled down robotic servicer to change out ORUs. In this configuration the RS would dock on a port in the center of GPS's +/- X panel (see above figure). Using one robotic manipulator arm with an OTCM end effector the RS could swap ORUs from its pallet to the ORU ports on the GPS S/V. The Scaled Down Ranger would need one dexterous arm (reduction factor of 50% for the arm subsystem). It would need a small docking port instead of a grapple arm (reduction factor of 75%). The video camera arm would be much shorter (reduction factor of 25%). Also, there would be only be one end effector verses Ranger's 7 end effectors (reduction factor of 75%). The mass breakdown is found in Spreadsheet #8.

The decrease in power, mass, and complexity of the payload will correspond to decreases in the supporting bus subsystems. First, with only one dexterous arm, no grapple arm, and easier ORU movements, the payload power requirement could be estimated to decrease by 50%. Since this is the majority of power requirements, we decreased the overall power by close to 50% (from 675 to 350 Watts). There are two reasons why the communications and data management could also be decreased by 50%. With much simpler tasks, fewer control variables, and a docking port on the bore-sight of the Robotic Servicer, we removed the bore-sight camera on Ranger. This should reduce the 4 Mbit / sec telemetry by approximately 50%. In addition, with no grapple or second dexterous arm the data processing and commanding requirements drop significantly. The one subsystem that does not reduce significantly is the Attitude Determination and Control System (ADCS). The reason is the ADCS still has to control both the RS and GPS. Since the GPS S/V mass does not change dramatically between the RS alternatives, the ADCS can only be reduced by 25%. These reductions can be see in Column 4 of spreadsheet #8. Those reductions are the only major difference between spreadsheet #7 and spreadsheet #8. Spreadsheet #8 is illustrated below.

Spreadsheet #8: Scaled Down Ranger (medium performance)

Notes:

Power: 350 W   Battery discharge energy = *430* W*hr

120 (AgZn)	2 yr (NiH-100% dis.)	15 yr (HiH-50% dis.)
5 kg	10 kg	19 kg

		pounds	% mass red.	kgs.	red. Factor (%)	med. Baseline	120 day	120 day (lbs.)	2 yrs.	2 yrs. (lbs.)	15 yrs.	15 yrs. (lbs.)
Spacecraft Bus		852	511	232		165	175	385	183	404	220	485
Data Management Systerr	14		8	4	*50*	2	2	4	2	5	3	6
Thermal	32		19	9	*50*	4	5	10	5	11	6	13
Communication System	30		18	8	*50*	4	4	10	5	10	5	12
ADACs & RCS	272		163	74	*25*	56	59	130	62	136	74	163
EPS (-solar arrays & batt.)	*121*		73	33	*50*	16	17	38	18	40	22	48
Sub Total =		469	281	128		82	87	192	92	202	110	242
Batteries	*78*						5	11	10	21	19	42
Solar arrays (silicon)	*50.6*				(Gal. Ars->)		7	15	8	18	10	22
Structures & Mechanism	251		151	68	*50*	34	34	75	35	77	36	80
Sub Total =		302					46	101	52	116	65	144
Payload		524	314	143		66	70	154	73	162	88	194
Dexterous Arms	232		139	63	*50*	32						
Grapple Arm	130		78	35	*75*	9						
Video Arm & Cameras	106		64	29	*25*	22						
End Effectors	56		34	15	*75*	4						
Total mass (in kgs.)		1376	826	374		188	203		217		263	

Figure 4.4-13. Summary of Scaled Down Ranger

4.4.8 Free Flying Robotic Servicer

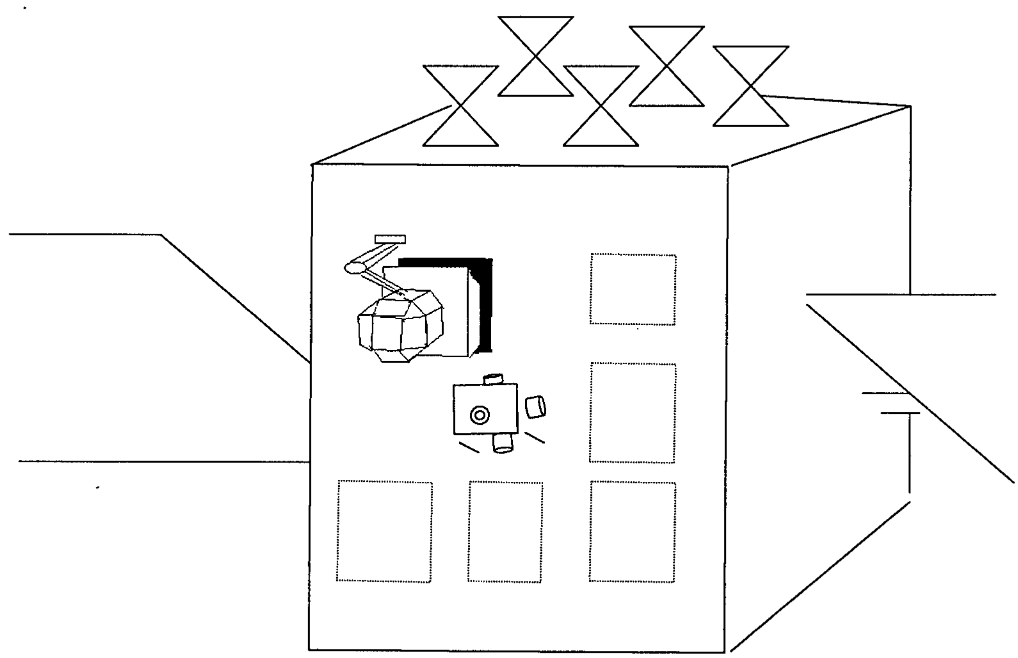


Figure 4.4-14. Diagram of Free Flying Servicer Concept

The concept for the free flying robotic servicer is in the RMS Analysis Procedure. The Servicing Micro-satellite (SMS) would be based on the micro-satellites being designed by Air Force Research Laboratory. One micro-satellite with very similar requirements to the SMS is XSS-10. This is a small self-contained satellite that will fly close to a target satellite and return videos of it. AFRL provided a detailed mass breakdown of XSS-10 (Madison, 1999). For costing purposes, we grouped those subsystems into three general categories. The categories are structures, electrical power system, and avionics. Since the SMS won't have the orbital maneuvering requirements, we removed the unibody engine and propellant subsystems from the XSS-10. The two subsystems that the SMS will require beyond the XSS-10 design are the OTCM and a robotic arm for the camera. The mass breakdown of the Free Flying RS is found in SS #9.

The RS transport vehicle (RTV) is very similar to the bus satellite of the Scaled Down Ranger (SDR). The batteries on the XSS-10 are 10 Amp*hr $LiSO_2$ batteries. Since AFRL did not provide me the voltage, we will assume the 28V aerospace standard. Therefore the stored energy requirement is 280W*hr. This requirement in addition to the bus requirements makes the battery requirements for the Free Flying RS similar to the SDR RS. However, the profile of power needs is somewhat different than the Scaled Down Ranger. On the SDR the nominal power is based on the requirements from the manipulators. While there will be fluctuations, the steady state power requirements during servicing should equal the power delivered from the solar arrays. For the Free Flyer the power is required to charge up the SMS before and not during the servicing mission. Thus the RTV's solar arrays only have to charge up the 280 W*hr SMS batteries. Therefore the total power requirements for the Free flying RS can be less. A complete analysis of electrical power profile would be necessary to determine the specific

reduction in power requirements. Since that analysis seemed beyond the conceptual scope of this study, we reduced the 350 Watts requirement to 250 Watts based on engineering judgement. However, the SMS might need to be recharged from the RTV during the "swapping" of old and new ORUs. Therefore we increased the battery discharge energy requirement of the RTV to 475 W*hr. On board the RS Transport vehicle there will be a propellant re-supply tank for the SMS. The amount of SMS propellant will be based on the number of servicings, which in turn is based on the Architecture. Hence, this subsystem is incorporated in the mass during the RS propulsion design versus incorporating it in this section. Spreadsheet #9 is illustrated below.

Spreadsheet #9: Free Flying Ranger: (Low Performance)

Notes:

Assume power needed: 250 W

Battery discharge energy = *475* W*hr

120 (AgZn)	2 yr (NiH-100% dis.	15 yr (HiH-50% dis.)
5 kg	11 kg	21 kg

		pounds	% mass red.	kgs.	red. Factor (%)	Baseline	120 day	120 days (lbs.)	2 yrs.	2 yrs. (lbs.)	15 yrs.	15 yrs. (lbs.)
Mothership		852	511	232		121	128	283	135	297	162	356
Data Management System	14		8	4	*50*	2	2	4	2	5	3	6
Thermal	32		19	9	*50*	4	5	10	5	11	6	13
Communication System	30		18	8	*50*	4	4	10	5	10	5	12
ADACs & RCS	272		163	74	*50*	37	39	86	41	91	49	109
EPS (-solar arrays & batt.)	*121*		73	33	*60*	13	14	31	15	32	18	39
Sub Total =		469	281	128		61	64	141	67	149	81	178
Batteries	*78*						5	12	11	23	21	47
Solar arrays (silicon)	*50.6*			(Gal. Ars->)		7	5	11	7	15	9	20
Structures & Mechanism	251		151	68	*50*	34	34	75	35	77	36	80
Microsat Dock						4	4	9	4	10	5	12
Sub Total =		302				45	43	96	46	102	51	111
Microsatellite				26		26	28	61	29	64	35	77
bus (XSS-10)						*18.5*	20	43	21	45	25	54
structures					*5*		5	12	6	12	7	15
EPS					*4*		4	9	4	10	5	12
avionics					*9.5*		10	22	11	23	13	28
camera arm (2.5 kg) + OTCM (5 kg)						*7.5*	8	18	8	18	10	22
				Total =		132	135		143		166	

Figure 4.4-15. Summary of Free Flying Ranger

4.4.9 Cost Modeling

Since costs are important to the user and can be estimated in many different ways, it was important to find a fairly accurate costing methodology. Another complicating factor in finding costs for robotic satellite servicers is there are no current operational servicers. Therefore, we used the NASA /Air Force Cost Model '96 Program (NAFCOM '96). NAFCOM '96 is a cost estimation computer program that uses weight relations to provide cost estimates of aerospace programs. It uses a work breakdown structure to give different estimation relations for the different spacecraft components. It is able to provide different cost estimations for individual components through its database of over 104 military and civil space programs. NAFCOM '96 was developed by Science Applications International Corporation for NASA's Marshall Space Flight Center and the Air Force Cost Analysis Agency.

The inputs to the NAFCOM models are the mass breakdowns of the different robotic servicers. Nine different servicers were calculated corresponding to the three performance ranges and three design lives (see Spreadsheets 7-9). NAFCOM required weight in lbs., so the spreadsheets convert the units. To make the cost process more efficient, the propulsion system cost was added later because it is dependent on the architecture type.

NAFCOM '96 provides three different methods for providing the cost relation for each component. The three methods are user define, specific analogy, and data base averages. For most components we used data base averages for the cost relation. In using data base averages we have to choose what type of component, for example antenna or electrical distribution subsystem. Then we had to select the database, for example all unmanned satellites, or only reconnaissance satellites, etc. For the robotic servicers we chose all earth-orbiting unmanned

satellites. The second method we used was specific analogy. For the manipulators of the robotic servicer we chose the specific cost relation to the mechanisms subsystem on Mars Pathfinder.

The final cost sheets are in spreadsheets 13 – 22 (Appendices I - R). We had NAFCOM use 1999 dollars with Air Force inflation figures. Our NAFCOM cost figures assume that a non-flight prototype test unit is built for each robotic servicer alternative. NAFCOM calculates the prototype unit cost as 130% of the first flight unit cost, and reports it in the Design, Develop, Test &Evaluate (DDT&E) category.

NAFCOM reports all its cost into two categories. The first category is the DDT&E, which is reported in my study as nonrecurring cost. The second category is production cost, which is reported as recurring costs in this study. In the case of alternatives with multiple robotic servicers, we calculated the cost for six robotic servicers. For alternatives with three robotic servicers we halved the production costs. Since the first three units cost more to produce than the next three, this statement is not quite true. However, the difference was not significant for this level of study. For example, the 120-day low performance robotic servicer the average production cost / unit was $8.5 M for six servicers and $9.2 M for three servicers.

In addition to the prototype and flight hardware cost, NAFCOM calculates the system integration costs. These costs include the following: the integration, assembly, and checkout, the system test operations, the ground support equipment, systems engineering, launch & operations support, and program management.

To verify NAFCOM's estimates of the robotic servicers costs, we compared it with estimates given by University of Maryland. NASA has contributed a total of $12+ million in U. of M.'s space robotics program. However, with two programs (TFX and TSX) and the basic research involved, Joe Parish estimated it would have cost $8 million to build TFX. He and

Gardell Gefke estimated a commercial production of an operational Ranger would be approximately double that cost (Parish, 1998). This is still less than the $17 to $27 million (depending on design life) NAFCOM estimated for production of Ranger. In addition, NAFCOM estimated $77 to $111 million for development cost. A summary of all the different Robotic Servicer costs is listed below. The ion and solar thermal propulsion costs are already calculated in the servicers costs.

	Free Flying Servicer: Standard	Free Flying Servicer: w/ ion or sol. th. Prop.	Medium Servicer: Standard	Medium Servicer: w/ ion prop. (Alt G)	Medium Servicer: w/ sol. Th. (Alt E & H)	Ranger Servicer: Standard
120 days						
Production cost/ unit	8.5		13.5			17.3
RDT&E	44.6		56.5			77.6
2 years		Alt. "G"			Alt. "H"	
Production cost/ unit	8.1	16.4	13.3	21.6	18	22.3
RDT&E	42.5	70.5	55.8	83.8	77.8	92.2
15 years		Alt. "E"			Alt. "E"	
Production cost/ unit	10	15	15.3		20	27.1
RDT&E	52.5	74.5	64.6		86.6	111.4

Ion propulsion Costs		Solar Thermal Costs	
RDT&E	28	RDT&E	22
Production	8.3	Production	4.7

Figure 4.4-16. Servicing Cost (Millions [$1999])

4.4.10 Simulation

The purpose of the simulation was to simulate alternatives with considerable uncertainty in their performance. We used Excel spreadsheets to generate performance numbers for alternatives that only involved upgrading the constellation. For repair alternatives (the C, D, E and F architectures), we used AweSim simulations to include the randomness of satellite

component failures in our performance assessment. The associated control file and network for each the four architectures are in alphabetical order by architectures in Appendices T, U, V, and W.

The simulation has three main loops. The first loop models the functioning of each satellite in the constellation. The second loop models a regular review of the constellation and corresponding repair decisions. The third loop collects repair statistics after repair operations have concluded.

4.4.10.1 First Loop – Overview

The first loop begins by generating entities to represent each satellite. The simulation creates the satellites in accordance with the current proposed launch schedule for the IIF constellation. Our sponsor provided us with this data. The simulation clones each of these satellite entities. Clone one helps track the number of IIF satellites active at any given time in the simulation. Clone two represents unrepairable failures on the satellite. The remaining clones represent repairable component failures. Each of those repairable clones enters queues and wait for availability of a servicer. When a servicer becomes available, if that satellite has not already suffered an unrepairable failure, the satellite receives service.

4.4.10.2 Second Loop – Overview

The review loop determines the location of the available servicer or servicers and considers which planes have satellites in need of service. The simulation then commits the servicers to conduct servicing missions.

4.4.10.3 Third Loop – Overview

After the last unrepairable satellite failure occurs, this loop collects statistics about the servicing missions.

Appendix S contains a detailed description of the AweSim model.

4.4.10.4 Output Analysis

The primary parameter of interest that the simulation provided was mean time to repair. The important contribution of the simulation was the ability to distinguish between alternatives in the context of this parameter. We performed twenty runs of each architecture category, and this produced mean times to repair and standard deviations. Two standard deviations above and below the mean captures approximately a 95% confidence interval over the actual mean of the data. If two intervals do not overlap, one can say that the means are statistically different. If there is overlap, the means are indistinguishable. The following table displays the means and 95% confidence bounds for each architecture category. The values are in months.

Table 4.4-6. Output Analysis

Category	95% Lower Bound	Mean	95% Upper Bound
Architecture C	5.599	6.165	6.731
Architecture D	0.123	0.287	0.451
Architecture E	2.343	2.699	3.055
Architecture F	2.302	2.678	3.054

The intervals only overlap for the E and F architectures. Thus, our use of the means as representative values for the corresponding architecture category was acceptable.

4.4.10.5 Verification and Validation

4.4.10.5.1 Verification

The question of verification and validation arises whenever a situation warrants use of a simulation model. According to Banks, Carson and Nelson, verification pertains to proper performance of the computer program (p. 16). We developed the simulations in small steps and conducted verification at each point along the way. Output data files verified that the model was accomplishing the tasks for which we designed it. This verification occurred whenever we made

modifications to the simulation. This iterative verification process proved quite useful, and would have been much more challenging if we had waited to perform verification until we felt the model was complete. The number of entities created in the simulation was important to the success of the verification efforts. We created and collected data on 24 entities. Each entity was cloned into three entities early in the simulation, and each clone was responsible for different properties of the original entity. The clones shared certain qualities and differed in others. Through the use of output data files, we monitored the relationships among the clones of each entity and could verify that the simulation was operating as it was intended to operate. An important benefit of the number of entities we created in the system was that it allowed the system to reach a steady state condition. With fewer entities, we would have missed some important interactions that could have impaired later simulation results.

4.4.10.5.2 Validation

Banks, Carson and Nelson define validation as "the determination that a model is an accurate representation of the real system" (p. 16). An analyst accomplishes this by comparing the simulation results to the outputs of the system he or she is trying to model. This was impossible because the majority of my simulation work involved hypothetical alternative architectures. The only validation came with regard to the failure mode distributions we chose. The failure distributions for random failures, the solar arrays, and the clocks were from Dr. Jim Womack's paper entitled Revised Block II/IIA Lifetime Predictions and the Impact on Block IIR/IIF Replenishment Planning (Womack, 1998). His paper took previous Block II/IIA reliability models and updated them with historical GPS data. Dr. Womack gave special attention to solar array degradation and clock reliabilities within that redundant system. That data helped us to develop a representative model for any constellation of GPS satellites. Dr.

Womack's predictions provided validation of the simulation results. In both cases, the predictions were for additional years of life remaining on the GPS satellites, and they concur with previous failure rates. However, the satellites about which we were most concerned were the Block IIF satellites. We were not able to acquire the necessary data to update the simulation. Thus, data collection and further validation are left to future users of this research.

4.4.11 Aerospace's Study

The GPS program office sponsored Aerospace Corp. to perform a companion study with our study. Since Aerospace has extensive experience with design of the GPS satellite, we requested they study the impacts of making the GPS S/V serviceable. The two important impacts to the GPS S/V is mass and cost increases. The mass increases are important because cost would sharply increase if the S/V overgrows its current launch vehicle's capacity. Cost increases are important because every GPS satellite would have the additional cost.

The mass and cost increases will come from two sources. The first source is the actual mechanical and electrical interfaces with the servicer and ORUs. The second source is from the enlargement of GPS's bus subsystems to provide the extra power, communication, and attitude control from additional payloads.

Aerospace's study is on-going. A status of their preliminary findings is in Appendix Y. Appendix Y has only provided impacts for upgrade capability, since that is the primary concern of our sponsor. It is interesting to note that the robotic servicer's performance did not significantly change the mass of the GPS S/V. However, without a full assessment, this observation could be misleading to the overall impacts, since there is no quantification of the level of servicing between robotic servicers. For example, a robotic manipulator would probably be able to service GPS antennas. A Free Flyer Servicer would not have this capability.

A second preliminary trade study Aerospace performed is the mass impacts due to the size of the ORU upgrade. Here the mass impacts to the GPS bus subsystem were evident. For the upgrade compartment approach the mass impacts were the following:

- Baseline GPS: 2813 lbs. (Wet mass)
- 50 kg upgrade: 3062 lbs.
- 150 kg upgrade: 3346 lbs.
- 300 kg upgrade: 3666 lbs.
  (Hall, 1999)

These values were based on the assumption that additional electrical power was needed. Another approach is to replace the existing payload. This approach requires much less additional bus subsystem support. The impacts for this approach would be the following:

- Baseline GPS: 2813 lbs. (Wet mass)
- 50 kg upgrade: 3007 lbs.
- 150 kg upgrade: 3165 lbs.
- 300 kg upgrade: 3287 lbs.
  (Hall, 1999)

The calculations for the 704 kg lift margin for the IIF / EELV medium combination was found in the Piggyback concept (Section 4.4.3.3). The largest S/V mass change in the above examples was 388 kg (853 lbs.). Therefore given the current system, the additional mass impacts should not be a significant issue. Cost data was not available at the time. As Aerospace's report becomes final, this would be an excellent area for future study.

4.5 Synthesize Systems into Alternative Solutions

4.5.1 Overview

Our process used a systematic way to design multiple alternatives to solve the problem. Much of the alternative designing was based on decomposing the requirements into a hierarchical system. We accomplished this in the previous steps where we analyzed each of the

four major components in detail. The four major components were orbital architectures, logistics and transportation system, the RS propulsion system, and the robotic manipulating system.

The next step was to synthesize the different options from each component into workable alternatives. The design space for the feasible alternatives was quite large. There were eight different orbital architectures, 21 launch vehicle and upper stage combinations, with 3 different ORU capacities, 3 different Robotic Servicers, and 2 different GPS constellation configurations (see Section 5.3). This resulted in 3024 different feasible combinations, which did not include minor deviations like flying the ORUs with the robotic servicer or piggybacking a servicing mission with a GPS S/V.

Analyzing all the different combinations was beyond the scope of this study. We chose a subset with the goal of representing a broad spectrum of servicing systems. The subset spans from a one-time, low performance servicer to a permanent, recurring, high performance servicer with 300 kg of ORU capacity. Additionally, we chose the subset such that it would address the three employment strategies we outlined in Section 4.3. For example, the architecture with quick response repair utilized the high performance servicer that could repair the largest percentage of the GPS S/V. Finally, the subset represented the most likely requirements. An example requirements guideline was that the alternatives with plane changing servicers utilized only the small or medium size robotic servicers.

4.5.2 Framework

To generate a consistent comparison, we used a baseline scenario to analyze the alternatives. In conjunction with our sponsor, we chose fifteen years of on-orbit servicing as the timeline. This coincided with the design life of the Block IIF constellation as the first potential users of on-orbit servicing. In this spectrum we chose two types of Robotic Servicing Systems:

the one time upgrade and the recurring servicing throughout a fifteen-year span. The recurring servicing scenarios had four upgrade servicing missions at year 1, 5, 10, and 15. The two architectures with scheduled repair (Alternatives E and F) had 7 servicing missions: 4 as combined upgrade and repair and 3 repair only missions. The repair only missions were at years 2.5, 7.5, and 12.5.

4.5.3 Process

Alternative generation synthesizes all the components that we have modeled. The synthesis process is in Figure 4.5-1 below.

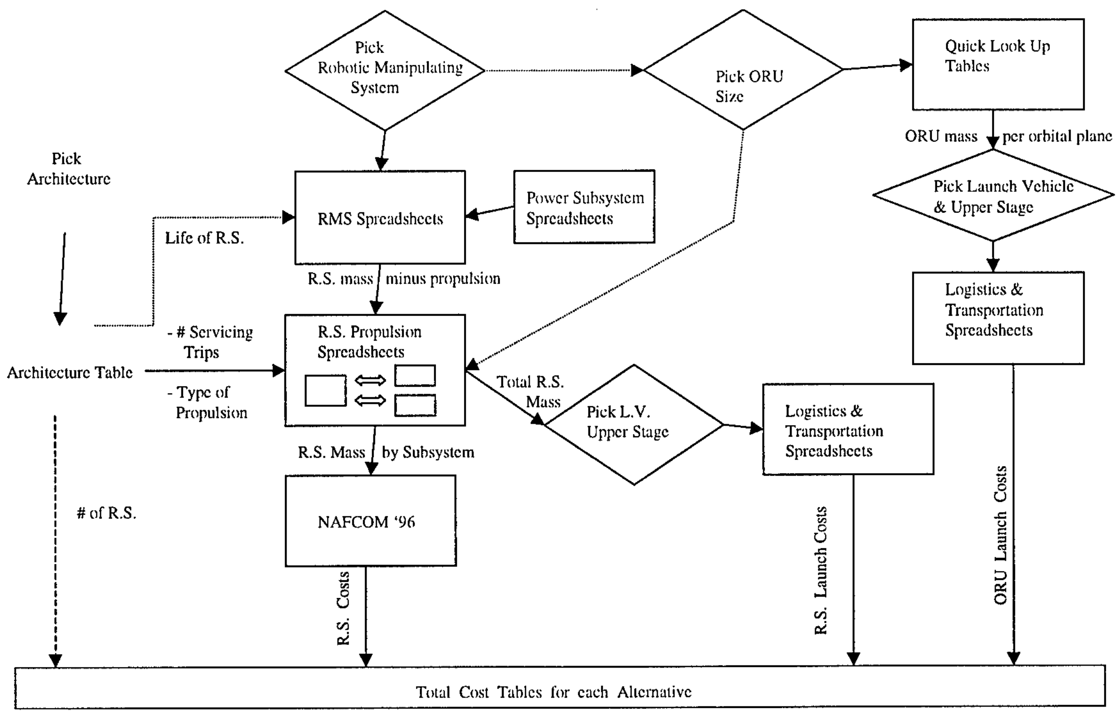


Figure 4.5-1: Flow Chart for Analyzing Cost for Each Alternative

The Orbital Architectures spanned the spectrum of requirements and provided the overarching concepts for servicing. For this reason, they represented the foundation by which we generated the alternatives. We modeled all eight architectures at least once, and we modeled the ones with desirable characteristics with up to ten different configurations. We transferred the

number of servicing trips and the type of propulsion into the RS Propulsion spreadsheets. The total cost tables below contain the number of robotic servicers. The life of the robotic servicer went into the robotic manipulation system (RMS) spreadsheets.

Next, we chose the type of robotic servicer. The RMS Spreadsheets generated the total robotic servicer mass minus the propulsion unit. We fed this value into the RS propulsion spreadsheets. In addition, we chose the ORU size for each alternative, and this was also an input into the propulsion spreadsheets. These spreadsheets determined the propulsion, propellant, and total mass of the robotic servicer. By subsystem, we inputted the RS mass into the NASA / Air Force 1996 Cost Model to determine total cost of the robotic servicer for each alternative. These costs are in the Summary Tables for Robotic Servicer Cost (Table 4.4.1). The RS Propulsion Spreadsheets also determine the total RS masses, which are summarized in the following table:

Mass Totals (in kg.)

High Performance Robotic Servicer

3-Plane

Alternative	A (or C)		G	B (or C) (D profile)			D (7 servicings)
Mission Life	120 day		2 years	15 years			
RMS + RS Bus	378		405	494			
ORU Mass	150	300		50	150	300	150 (upgrade) 20 (repair)
R.S. Prop	26	26		36	36	36	36
RS with Propulsion	562	611		738	818	937	1021

6-Plane

ORU Mass	150	300	300	50	150	150 (upgrade) 20 (repair)
R.S. Prop	26		320		36	36
RS with Propulsion	470		3231		639	715
			LEO --------> 3868			

Figure 4.5-2. High Performance Servicer Mass Totals

Medium Performance Robotic Servicer

3-Plane

Alternative	A		G	B (or C) (D profile)		E
Mission Life	120 day		2 years	15 years		
RMS + RS Bus	203		217	263		
ORU Mass	150	300		50	150 (up.) 20 (repair)	150 (up.) 20 (repair)
R.S. Prop	14	14		18	18	190 (engine) + 170 (tank)
RS with Propulsion	360	487		412	491	623
						refuel = 11,470

6-Plane

ORU Mass	150	150	300	300
R.S. Prop	14	320	320	18
RS with Propulsion	276	1922	3000	387
	LEO ---->	2302	3591	

Figure 4.5-3. Medium Performance Servicer Mass Totals

Low Performance Robotic Servicer

3-Plane

Alternative	A	B
Mission Life		15 years
RMS + RS Bus	138	166
ORU Mass	150	150
R.S. Prop	10	12
RS with Propulsion	266	318
Free Flier Propellant	26	26 mult 4 = 104 kg.
R.S. Total	292	422

6-Plane

Alternative	A (with ORU)		G	B	F	E
Mission Life	120 day		2 years	15 years		
RMS + RS Bus	138		143	166		
ORU Mass	50	50	50	50	50 (upgrade 20 (repair)	50 (upgrade) 20 (repair)
R.S. Prop	10	10	146	12	23 + 130 (tank)	190 + 130 (tank)
RS with Propulsion	170	370	289	235	280	486
Free Flier Propellant	13		79	13 mult 4 = 52	20	20
R.S. Total	183	378	864	287	300	506
			LEO = 1034		8,671 kg. (refuel)	8,745 kg. (refuel)

Figure 4.5-4. Low Performance Servicer Mass Totals

We chose the required launch vehicle and upper stage for the RS based on its total mass. While there may be additional criteria in choosing a launch vehicle, the primary motive is to keep launch cost down. With this goal, we chose the lowest cost launch vehicle and upper stage with the required lift capability. Using the LTS Spreadsheets we determined the launch vehicle, upper stages, and dispenser recurring and nonrecurring cost. These cost are also reflected in the Total Cost Tables.

The next step was to determine the cost of transporting the ORUs to the robotic servicers. Using the Quick Lookup Tables (spreadsheet #6) we were able to determine the ORU mass for each orbital plane. Using this value, we chose a launch vehicle based on the same criteria as the robotic servicer process. Again using the LTS spreadsheets we were able to determine the recurring and nonrecurring cost for transportation of the ORU.

4.5.4 Table of Overall Cost and Performance

Two types of outputs resulted from designing each of the alternatives. The first outputs were the intermediate design parameters needed to characterize the alternative. Some of these included physical characteristics of the RSS components, orbits used, and mass of the components. The second type were output variables that will be used to evaluate the alternative versus the user's value system. Examples of these variables were the times for the different segments, the RSS's ORU capacity, and cost of the different components

Referring to the Table of Overall Cost and Performance (Appendix Z), it was apparent that there were many different configurations and associated cost. The top four lines represent input variables defining the concept definition of each alternative. The next five lines are determined variables based on the process described above. The next three lines represent the chosen launch vehicle and upper stage combination for each alternative. The next 5 lines

represent the individual components of recurring cost. The next line is the total cost for the first upgrade mission. If the alternative has recurring servicing missions, the next line is the total for all the servicing missions in the 15 year time span. See the Framework section above for details on the frequency of servicing missions during the 15 years. The average cost / mission is the total recurring cost divided by the number of servicing missions. The next three lines represent the individual components of nonrecurring cost. This is totaled in the fourth line under RDT&E. The final three lines are the total mission cost. The first line is the total mission cost for the first mission, which is the sum of the first mission recurring cost and the RDT&E cost. The second line is the total cost for each alternative. This value is the total recurring cost plus nonrecurring cost. The final value is the total cost per mission. This amortizes the RDT&E cost over all the servicing missions.

4.5.5 Alternative Generation Results

As alternatives were being generated, Architectures "A" and "B" seemed to provide low cycle time, and high capability, and cost for a relatively low cost. This was a subjective observation that will be assessed in the evaluation step. However, for this reason we exerted additional effort in generating many alternatives of those architectures. The first nine "A" alternatives are different combinations of RS performance and ORU size. The tenth alternative was different in that it launched the RS and ORUs on one mission. Because of the size of the GPS constellation, this could only be done with the low performance servicer and small (50 kg) ORUs. The eleventh alternative was like the tenth alternative except it piggybacked on six GPS S/V launches instead of requiring one dedicated launch. The drawback was that with a GPS Block IIF launch rate of two per year, this required three years for on-orbit cycle time. The advantage is with no launch costs this represented the lowest cost alternative. The "B"

alternatives were also different combinations of robotic servicing performance and ORU size. As one could see in the total cost, by only launching the robotic servicers once, the average mission cost was lower than the "A" alternatives.

Architecture C had the same robotic servicing architecture as "A" or "B". The difference between Architecture C and "A" or "B" architectures was "C" included scheduled repair during an upgrade servicing mission. Architectures A or B assumed the servicer only upgrades the GPS S/V. This difference changed the GPS S/V configuration but was transparent to the Robotic Servicing System. One assumption was the total ORU mass inserted on the GPS S/V is the same. We did not include separate Architecture C alternatives because the "A" and "B" alternatives can also be "C" alternatives.

The "D" alternatives represented an upgrade and fast repair combination. These were similar to the "B" alternatives except for the mini-depot in each orbital plane, and extra fuel for the servicings. Since quick response repair would demand flexibility from the servicer, we modeled these two alternatives with high performance servicers.

The "E" and "F" alternatives represented the precessing on-orbit depot. The total cost sheet shows that even with low performance and ORU capacity, these were expensive alternatives. Since these architectures were not better in other value categories like cycle time, we did not generate many of these alternatives.

The "G" and "H" architectures were servicers with direct orbital plane change capability. Since maneuvering between all the planes required approximately 10 Km/s delta V, the mass of the robotic servicer played an important role in overall cost. For this reason, we explored the combinations of low and medium performance servicers. Size of ORUs had much less impact on cost because they were not onboard the robotic servicers when they had to make plane changes.

Alternatives "G" and "H" had high performance propulsion systems. For this reason, it was more cost effective to launch them into Low Earth Orbit (LEO) and use their own propulsion systems to maneuver into the GPS orbit.

The following figure illustrates the cost differences between the different alternatives. The first mission includes all the develop cost and any additional cost the 1st mission has over the rest of the mission. For example, most of the alternatives would launch robotic servicers on the first mission, and would only launch ORUs on subsequent missions. The mission average represents the long term average cost of that alternative. Recurring cost represents the cost of a standard mission once the servicing alternative is operational. High, medium, and low capacity refers the size of ORUs used for servicing. The "H","M","L" letters underneath the table represent the high, medium, and low robotic servicers.

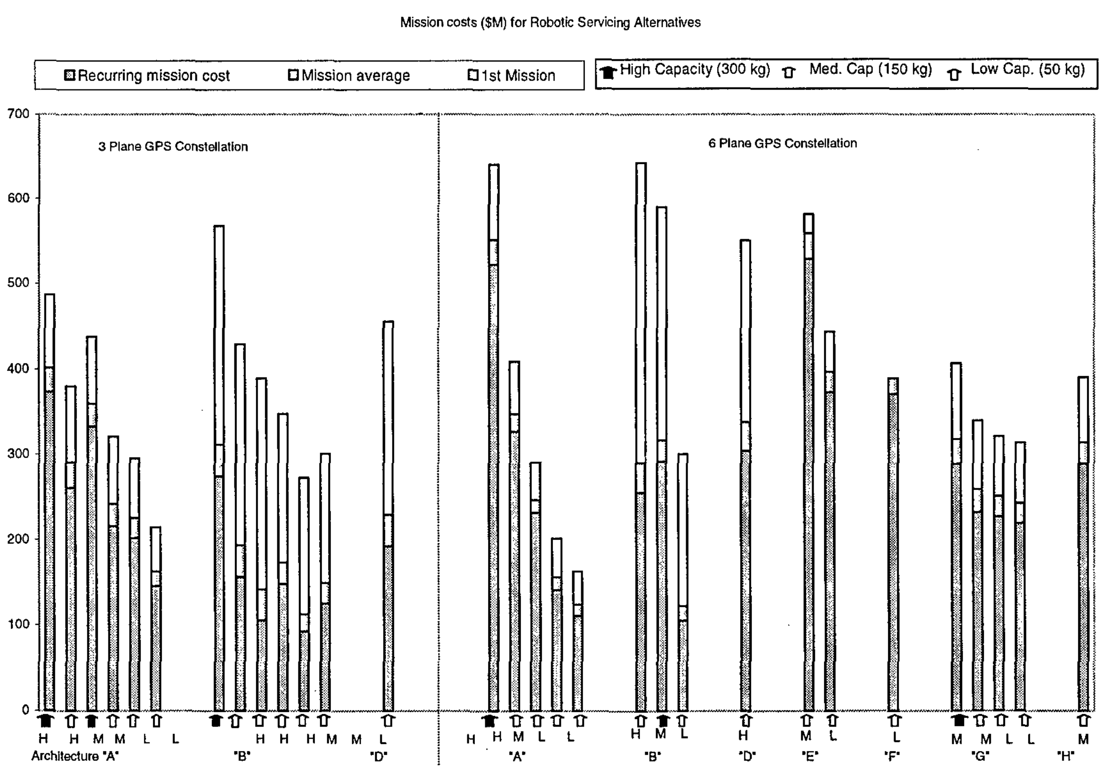


Figure 4.5-5. First Mission Costs and Average Mission Costs

While we will more fully analyze the cost data in the decision analysis portion of our study, we can draw some general conclusions. By looking at Architectures A, B, and D, we can see that on the whole, a 3-plane constellation will have slightly lower alternative costs. The major anomalies are the last two 'A' alternatives which had combined ORU / RS launches or were piggybacked on a GPS S/V launch. We did not explore these concepts with 3-plane configurations, so no comparison is available. Another noticeable characteristic is that the 'A' alternatives have much higher recurring costs than 'B' alternatives. This also makes sense, since 'B' alternatives launch a long life servicer on the first mission, whereas 'A' alternatives launch a robotic servicer for every servicing mission. A final feature to notice is that the architectures with RS's in every orbital plane have costs comparable to the architectures with one servicer.

4.6 Assess Value Functions

Each measure had a value function to convert scores to values. See Figure 4.2-1 for the value hierarchy. We assessed the functions from our sponsor contact, Howard Wishner. We presented the overall value model to Col Miller, the CZS commander and Howard's boss, and he accepted our work. We used a piecewise linear function for each measure. The functions all had an adequate range to include the score for the current baseline GPS architecture. The figures below are the plots of the value functions.

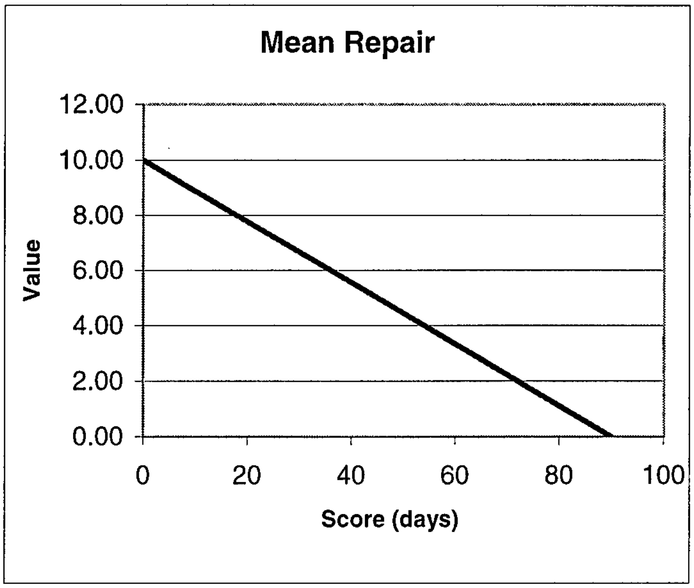


Figure 4.6-1. Mean Repair Value Function

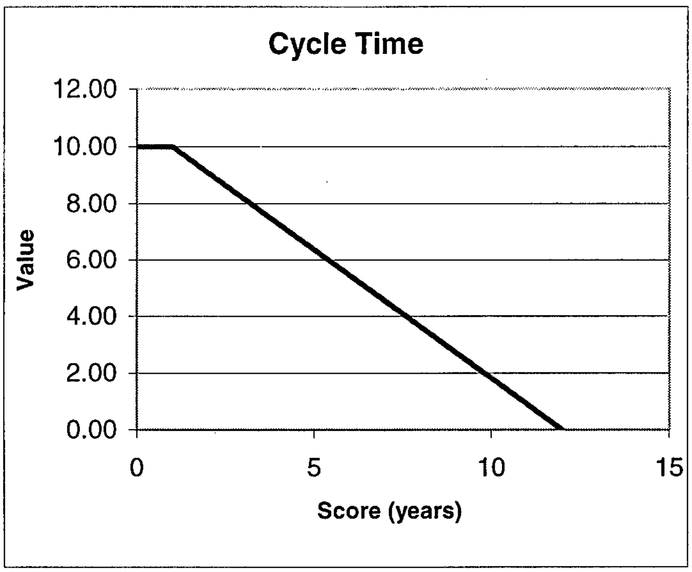


Figure 4.6-2. Cycle Time Value Function

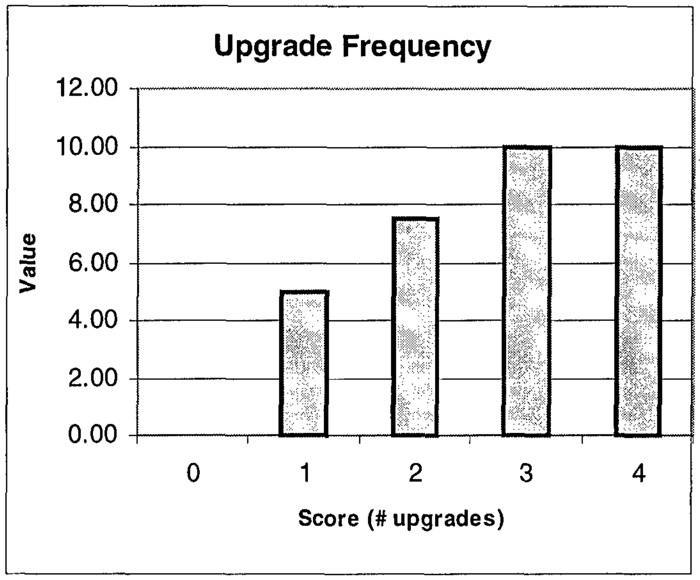


Figure 4.6-3. Upgrade Frequency Value Function

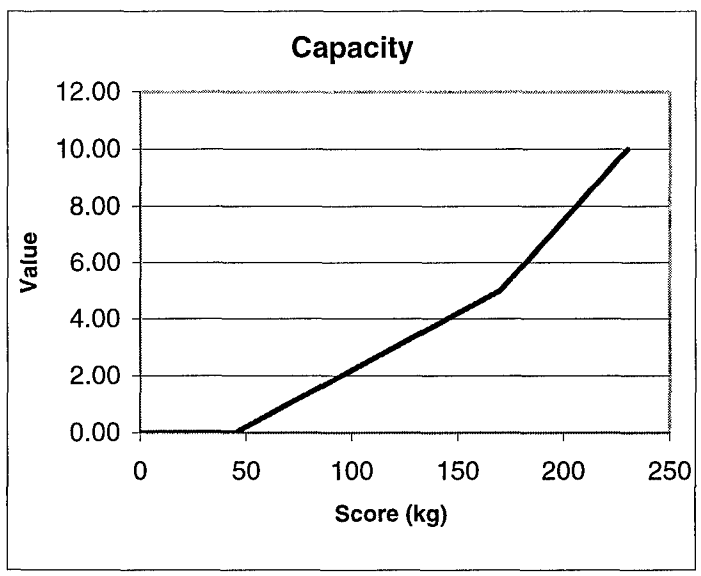


Figure 4.6-4. Capacity Value Function

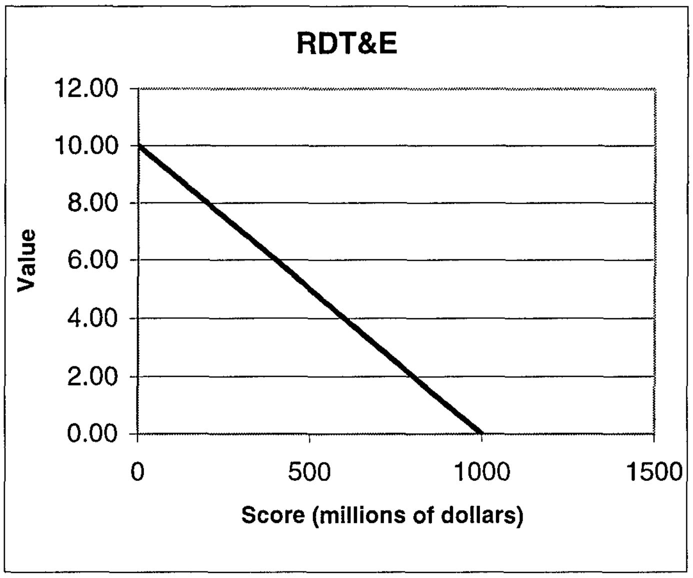


Figure 4.6-5. RDT&E Value Function

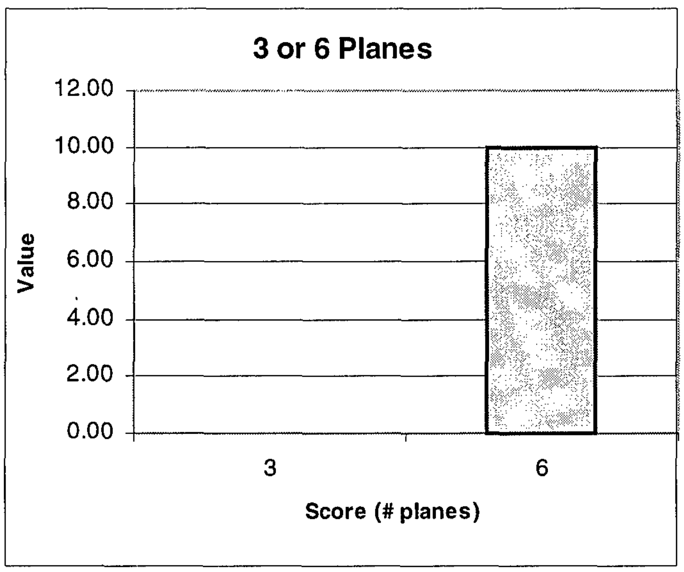


Figure 4.6-6. 3 or 6 Planes Value Function

The Multi-usability evaluation measure used a constructed scale. The possible levels were 0, Low, Medium and High. The 0 level corresponded to no servicer, and could not be used on other satellite programs. The Low Multi-usability level corresponds to the low capability servicer, the Medium level corresponds to the medium capability servicer, and the High level

corresponds to the high capability servicer. As servicer capability increases, so does flexibility. Thus, the direct correlation between Multi-usability and servicer capability was a good method for defining the Multi-usability measurement scale. The following figure is the Multi-usability value function.

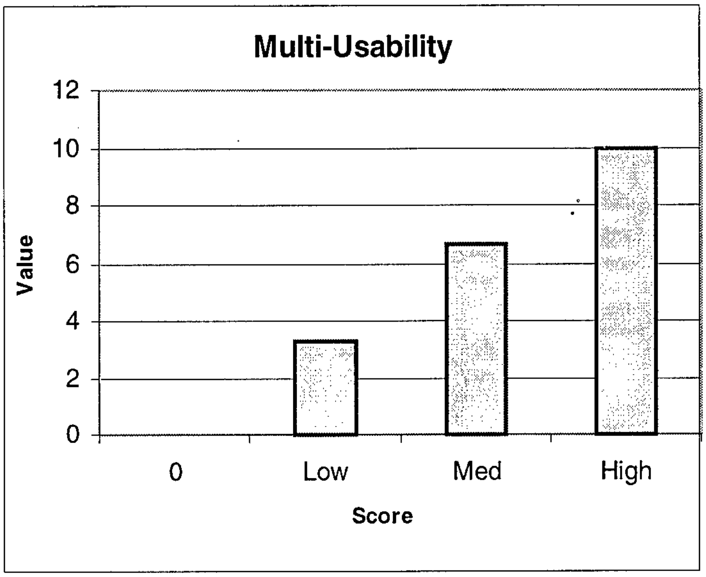


Figure 4.6-7. Multi-Usability Value Function

Orbit Transfer Capability was also a constructed scale measure. It had three levels: 0, Phase (P), and Phase and Transfer (P&T). A score of 0 indicated no phasing or orbit transfer capability. The Phase score applied to alternatives where the servicer can only perform phasing. Phasing was the ability to move between satellites in the same orbit. Alternatives that warranted the Phase and Transfer score could perform both functions. Transfer was the ability of a servicer to change from one orbital inclination to another. The figure below is the Orbit Transfer Capability value function.

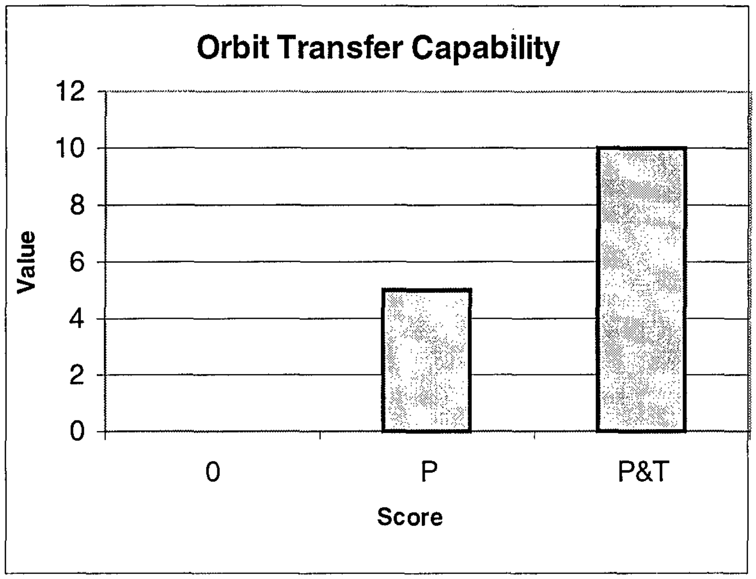


Figure 4.6-8. Orbit Transfer Capability Value Function

4.7 Assess Weights

Using the procedure that we outlined in Chapter 3, we assessed weights for the various measures. We set the Cycle Time measure as the baseline weight. We then asked our decision-maker, Howard Wishner, for the impact of each measure relative to Cycle Time. The following table shows the relative impact of all the measures. For instance, Mean Time to Repair has a relative impact of 0.25. Thus, relative to Upgrade Frequency, it has 0.25/0.50 = 0.50 the impact on the decision-making process as Upgrade Frequency. The numbers in the third column are the results of summing these impacts to one and solving for their individual weights. To demonstrate this calculation, consider the Mean Time to Repair and Upgrade Frequency information above. If those two were the only measures in the value model, we could calculate their weights as follows.

*Weight (Mean Time to Repair) = 0.50 * Weight (Upgrade Frequency)*

Weight (Upgrade Frequency) + Weight (Mean Time to Repair) = 1

*Weight (Upgrade Frequency) + 0.50 * Weight (Upgrade Frequency) = 1*

Weight (Upgrade Frequency) = 1/1.5 = 0.667

Weight (Mean Time to Repair = 1 – 0.667 = 0.333

Table 4.7-1: Measure Weights

Measure	Relative Impact	Weight
Cycle Time	1	0.190476
Shared RDT&E	1	0.190476
3 or 6 Planes	0.75	0.142857
Capacity	0.75	0.142857
Multi-Usability	0.75	0.142857
Upgrade Frequency	0.5	0.095238
Mean Time to Repair	0.25	0.047619
Orbit Transfer Capability	0.25	0.047619

The table above is sorted by weight and provides a valuable insight into the decision-maker's thought process. The overall weight under the first-tier evaluation consideration of performance was roughly equal to the overall weight for program viability. Thus, the relative impacts of these two areas on his decision-making were very close, which went against our intuition that performance was most important. This knowledge could impact future alternative generation efforts.

4.8 Evaluate Alternatives

4.8.1 Overall Results

Two primary goals of this research were to identify the highest performing alternatives and to understand the influence on that performance of the alternatives' many features. The following table is the final overall value score of each alternative in rank order. See Appendix Z for the details of each alternative.

Table 4.8-1. Overall Value Scores

Rank	Alternative	Value
1	Alternative 22	8.8736296
2	Alternative 16	8.6169524
3	Alternative 1	8.5838095
4	Alternative 13	8.196381
5	Alternative 26	7.9772641
6	Alternative 23	7.91114
7	Alternative 12	7.576381
8	Alternative 30	7.5385628
9	Alternative 21	7.4250582
10	Alternative 5	7.347619
11	Alternative 27	7.3124156
12	Alternative 2	7.1647619
13	Alternative 25	6.9659885
14	Alternative 24	6.9198932
15	Alternative 28	6.877619
16	Alternative 20	6.8061905
17	Alternative 14	6.7478095
18	Alternative 4	6.7038095
19	Alternative 29	6.3719913
20	Alternative 17	6.3598095
21	Alternative 9	6.3452381
22	Alternative 10	6.342381
23	Alternative 3	6.3266667
24	Alternative 15	6.1830476
25	Alternative 11	6.0114892
26	Alternative 19	5.8919048
27	Alternative 6	5.8752381
28	Alternative 18	5.8255238
29	Alternative 7	5.4214286
30	Alternative 8	4.8966667
31	Baseline	3.3333333

With the exception of the gaps between the three lowest performing alternatives, the gaps between overall value scores are quite small. The top alternative, Alternative 22, earned a score of 8.87 out of a possible 10. These results represent a broad spectrum of possible configurations and corresponding performance parameters. It is also important to remember that this list of 31 alternatives is not exhaustive. These alternatives are meant to be representative of the spectrum of possibilities. We were not able to enumerate all alternatives due to the length of time each alternative required for complete evaluation.

The closeness of the scores reminds us of something further. The performance parameters of these alternatives are theoretical. We calculated these values using sound engineering practices in conjunction with spreadsheets and simulation, but the only alternative that exists is the current Baseline alternative. Even if we had an alternative that shone above the rest, it would be important to assess the impact of variability in actual performance of the top alternatives. The method of assessing this potential variability extends to variabilities in the parameters of the model and is called sensitivity analysis. See the following section for that analysis.

4.8.2 Value Versus Cost Plot

Having determined the overall value and subsequent ranking of each alternative, it was finally time to bring in the cost information. The figure on the following page is a plot of value versus cost to GPS in millions of dollars. To add meaning to the plot, the symbols for each alternative represent three dimensions of the data for a total of five dimensions including the vertical and horizontal axes.

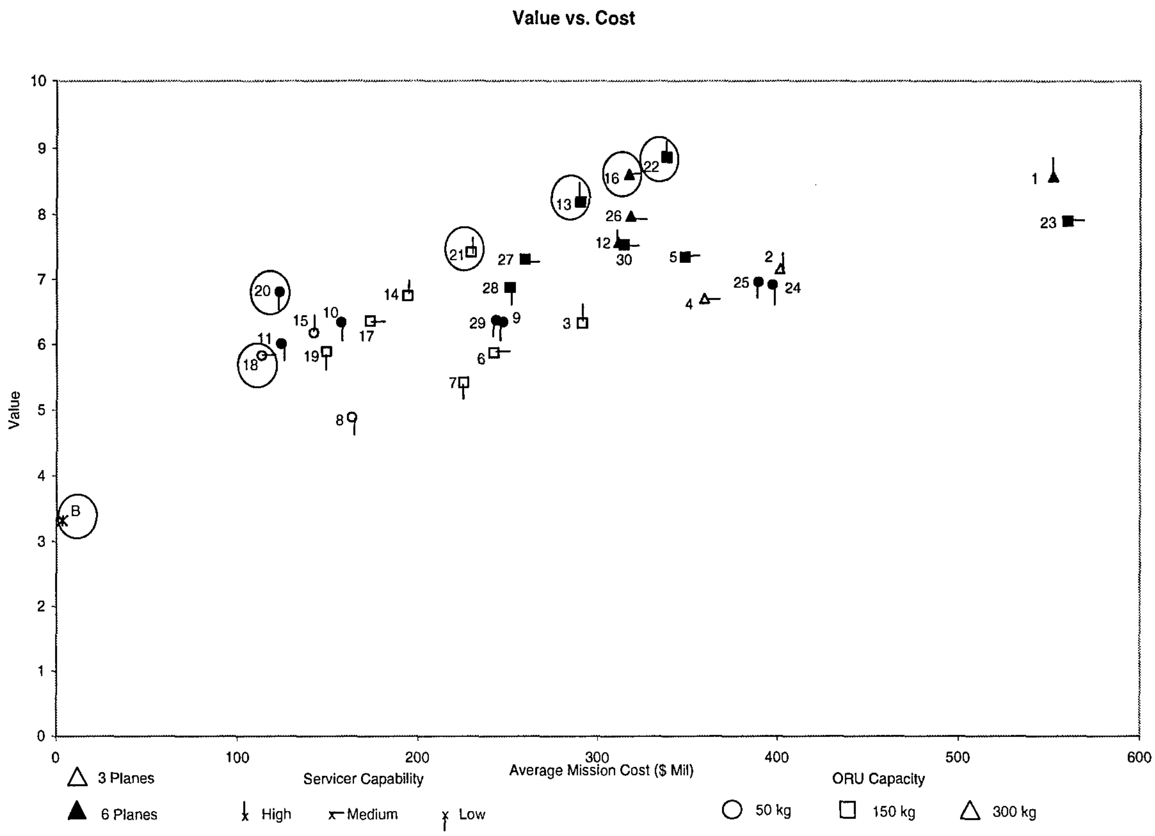


Figure 4.8-1. Multidimensional Plot of Overall Value vs. Average Mission Cost

The number by each symbol is the alternative number. The "B" symbol is for the Baseline alternative. The alternatives below and to the right of the circled alternatives earned less value and were more costly than at least one alternative among the circled ones. These circled alternatives, therefore, represent the set of best benefit to cost tradeoffs. The legend explains the other markings.

The purpose of combining so much information on one plot was to gain insight into the interactions of the various parameters and to understand which combinations were successful. By focusing on one set of symbols at a time, it was possible to isolate a dimension for the purpose of comparison. At the same time, the presence of the other dimensional information made it easy to take the more general view and examine potential interactions.

The cost axis in the value versus cost figure is average mission cost. Thus, we have used a long-term average mission cost as our basis of comparison. This does not reflect the actual cost of an alternative to GPS. Some alternatives perform only one servicing mission, and the servicer has a 120 day or 2 year design life. Other alternatives perform four servicing missions over a design life of 15 years. We chose this method of comparing value to cost, because we felt it was the most descriptive method to represent the data. We provided several costs that led up to average cost per mission in our spreadsheets, so that future users could focus on the data of interest to them.

4.8.3 Observations

The following observations reflect only the ranking of the alternatives by value. We drew conclusions with cost in mind in Section 4.8.5, "Value Versus Cost in Detail", below. By examining whether or not the marker is hollow, which indicate the alternatives' performance in the 3 or 6 Plane measure, one can see that the top six alternatives are six-plane variants. There was no attempt to estimate the cost to GPS for transitioning the constellation to three planes. We assessed that impact in the Program Viability portion of the hierarchy. However, looking to the right on the plot, we can see that six-plane alternatives can be significantly more costly than their three-plane counterparts. Alternatives 1 and 2 were similar in design with the exception of the number of planes. Alternative 1 was a six-plane design, and it exceeded Alternative 2 in both value and cost. The value differential is almost one and a half units, and the cost differential is more than 150 million dollars. Deciding whether such a tradeoff is justified is the decision-maker's responsibility. This situation highlights the importance of the value model as a tool and not a substitute for the decision-maker.

We made several additional observations from the plot. It is not until the thirteenth ranked alternative that an alternative has a low capacity servicer or a low capability servicer. Aside from the Baseline alternative, the bottom five performing alternatives were three-plane configurations. Alternative 18 was among these five, and it was the next least expensive alternative from the Baseline. These observations are important as indicators of the information the decision-maker can glean from the value model's results.

Three general statements came from these observations.

1. Six-plane alternatives appeared to outperform 3-plane alternatives
2. Medium and High capability servicer alternatives outperformed Low capability servicer alternatives.
3. Medium and High capacity servicer alternatives outperformed Low capability servicer alternatives.

A fourth point of interest was to determine if the choice of one servicer per plane or one servicer for the constellation had a significant impact on the results.

4.8.4 Statistical Analysis

The statistical method for testing these statements is to group the results according to the parameter of interest and then perform a pairwise comparison of their mean, or average, overall value. We then subject the difference between the two means to a T-test at a certain level of significance. We chose an alpha of 0.05. If the confidence interval about the mean contains zero, we cannot reject the hypothesis that the true means of the two groups are essentially the same. In other words, if the confidence interval contains zero, we cannot distinguish between the two groups in question, and we cannot say that one will generally performs better than the other does. However, if the confidence interval does not contain zero, with 95% confidence we reject the hypothesis that the two means are essentially the same. (Wackerly, 353) From that we can deduce that the group with the higher mean performance outperforms the other group.

We used the t-test, because it handles data for which we cannot apply large-sample techniques. There were three important assumptions that accompany the use of the T-test for comparing means. First, we assumed we randomly selected the data from a normal population. "This is appropriate for samples of any size and works satisfactorily even when the population is not normal, as long as the departure from normality is not excessive." (Wackerly, 1996: 353) Second, we assumed the two populations had a common but unknown variance. Third, we assumed the samples were independent.

The figure below is a plot of confidence intervals that test the above observations. The vertical line at zero on the horizontal axis helps the reader identify the statistically significant comparisons. As you can see, only three of the comparisons did not yield statistical significance.

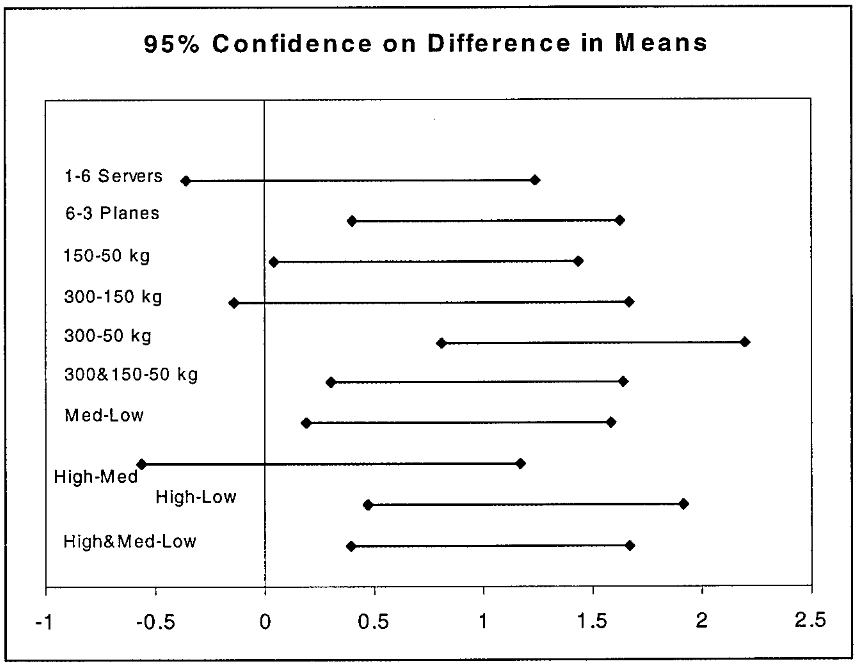


Figure 4.8-2. 95% Confidence on Difference in Means

The labels to the left of each confidence bar explain the elements that were involved in the test and the order in which we compared the means. For example, the first label, "1-6 Servers," explains the error bar to its right as a comparison of the alternatives grouped by number of

servicers. An alternative qualified for the 6-servicer group if it had one servicer per plane. We subtracted the mean overall value of alternatives in the 6-servicer group from the mean overall value of alternatives with 1 servicer. The "300&150-50 kg" error bar shows the confidence interval around subtraction of the mean overall value for 50 kg capacity servicer alternatives from the mean overall value of the combined 300 kg and 150 kg capacity servicer alternatives. The high, medium and low comparisons refer to our grouping of the alternatives according to servicer capability. See Appendix AA for the MathCAD 7 worksheet calculations.

We gained several valuable insights from the confidence intervals.

- 1 servicer for the constellation was no different from 1 servicer per plane.
- 6-plane alternatives, on average, performed better than 3-plane alternatives. This result was counter to our intuition that the increased efficiency of 3 planes would outweigh the negative aspects of transitioning the constellation from its current configuration.
- The 150 kg capacity alternatives outperformed the 50 kg alternatives by a narrow margin, and the 300 kg alternatives significantly outperformed the 50 kg group. This was a reflection of the decision-maker's perception that more mass per mission per satellite was better. We based our choice of masses on masses of existing systems. The impact of this alternative property may change as requirements and technology mature.
- The 300 kg and 150 kg alternatives were not statistically different. This validated our decision to group them together for the purpose of comparing them with the 50 kg alternatives.
- The capability comparisons followed the same pattern as the capacity comparisons. The medium, high, and medium/high capability groups all outperformed the low capability alternatives. The high capability servicer alternatives did not differ statistically from the medium capability alternatives. Thus, the low capability servicers, on average, yielded lower overall value than both medium and high capability alternatives.

Another insight came from examining the alternatives along the dashed line in Figure 4.8-1. All of these alternatives were capable of 4 servicing missions over a design life of 15 years. These observations and the accompanying analysis do not guarantee causal relationships. The multiple occurrences of significance, however, strongly suggest further investigation.

4.8.5 Value Versus Cost in Detail

We made further observations from the value versus cost plot. The general appearance of the data, and the specific trend of the circled alternatives, showed that an increase in cost will accompany an increase in value. Alternative 18 was the first alternative along the dashed line above the Baseline alternative. For $113 million, the user received a low mass capacity, medium capability servicing architecture designed for three planes. For an additional $10 million, Alternative 20 yielded a low capacity, low capability architecture for the existing 6-plane configuration. Alternative 21 was the next higher value circled alternative for $229 million. It utilized a three-plane configuration with a high capability and medium capacity servicer. The remaining alternatives were six-plane designs. At a total cost of $290 million, the next upgrade was alternative 13. Alternative 13 used a high capability servicer with medium capacity. Alternative 16 consisted of a medium capability and high capacity servicer for $317 million. Finally, the top alternative, number 22, cost $338 million and has a high capability and medium capacity servicer.

With a specific goal in mind for the overall performance of the GPS constellation, the decision-maker may trade value and features for flexibility. For instance, the cost to upgrade from Alternative 20 to Alternative 21 was an additional $106 million. The GPS JPO could select Alternative 20 and use that $106 million to improve the satellites' design or other dimensions of the constellation that were not in the scope of our analysis. The end result could be a constellation that outperforms Alternative 21 for the same cost. The decision-maker must take this information and determine what increase in value and change in features warrant the corresponding increase in cost.

4.9 Perform Sensitivity Analysis

4.9.1 Weight Sensitivity Analysis

Sensitivity analysis of the weights provided an understanding of the robustness of the weight structure. It was possible to accomplish this by taking one measure at a time and varying its weight across the full range from zero to one. As the previous section explained, the other weights would vary proportionately such that the total of the weights would still be one. The plots of overall value versus measure weight for the top alternatives revealed breakpoints where another alternative took first place. The following graphic provides a guide to the sensitivity analysis results for the weights.

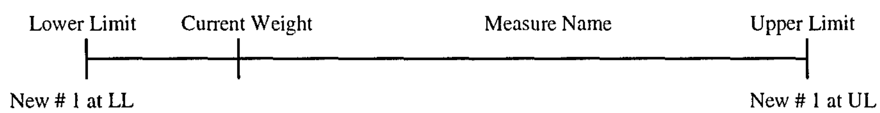


Figure 4.9-1. Weight Sensitivity Analysis Legend

The "Lower Limit" (LL) was the weight at which a new alternative becomes the best alternative as the weight decreases from the level the decision-maker chose. The "New #1 at LL" was the number of the alternative that takes the place of the previous best alternative. The "Upper Limit" was the weight at which a new alternative becomes the best alternative as the weight increases. The minimum possible weight was zero, and the maximum possible weight is one. If the Lower Limit was zero or the Upper Limit was one for a measure weight, the current best alternative remained the best alternative at that limit. The figure below shows the results of the one-dimensional sensitivity analysis of the weights.

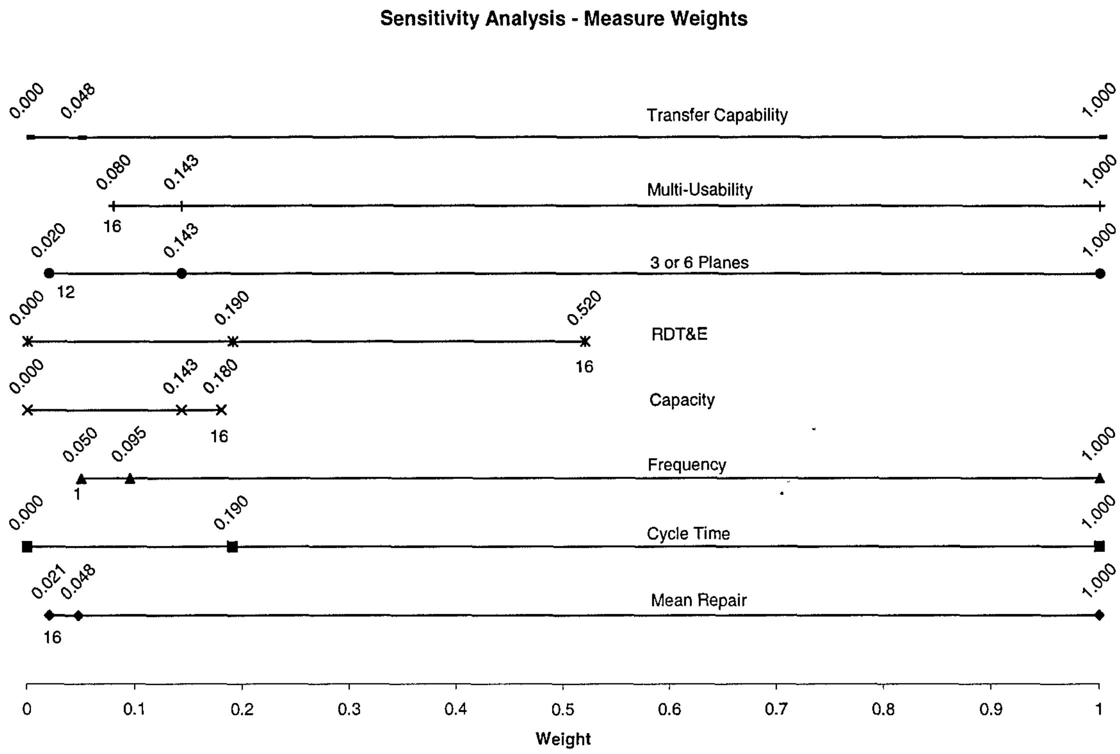


Figure 4.9-2. Weight Sensitivity Analysis Results

The above figure revealed several sensitivities of the results to the choice of weights. Multi-Usability, 3 or 6 Planes, Frequency and Mean Repair were sensitive as their weight decreased. RDT&E and Capacity were sensitive as their weight increased. The table below contains the absolute and relative changes that would lead to a policy change in the sensitive measures.

Table 4.9-1. Weight Sensitivity Analysis Thresholds

Measure	Absolute Decrease	Relative Decrease	Current Weight	Relative Increase	Absolute Increase
Mean Repair	0.0266	2.268	0.0476		
Frequency	0.0452	1.905	0.0952		
Capacity			0.1429	1.26	0.037
RDT&E			0.1905	2.73	0.33
3 or 6	0.123	7.143	0.1429		
Multi-Usability	0.063	1.786	0.1429		

In an absolute sense, the Mean Repair weight has the shortest distance to go before the choice of best alternative changes. However, because the weights were reflections of relative

impact between measures, it was more meaningful to determine the most significant relative weight changes that leads to changes in the decision. The smallest relative change in weight that would lead to a change in the ranking of the top alternative was with the Capacity measure. A 26% increase in the weight the decision-maker assessed for the Capacity measure would change the best alternative from Alternative 22 to Alternative 16. Changes ranging from 78% to 700% are required in the remaining measures to cause change. The 3 or 6 measure was the least sensitive of the sensitive weights, as the table above shows. This was also the measure that shows the most significant change in rankings at the decision change point. Alternative 12 became the top choice, and it was originally number seven in the rankings. The other changes in decision involved alternatives that were all from the original top three. Thus, the top three alternatives were fairly insensitive to fluctuations in the weight structure of the value model. This was useful because it meant, for this set of alternatives, the weights could potentially change to reflect the decision-making emphasis of another organization, and the current results would be robust to those changes.

4.9.1.1 Comments on Independence of Measures

After several iterations with our sponsor, we arrived at a complete value hierarchy. We were careful to avoid redundancy, and we believe we were successful. In fact, our efforts to avoid redundancy added a few iterations to the development of the hierarchy. The hierarchy was also operable and possessed small size. However, we were not able to fully achieve independence among the alternatives. The multi-usability (capability level) of the servicer dictated a bulk of the RDT&E cost. There was a negative correlation between orbit transfer capability and cycle time. Orbit transfer capability reflected a design with one servicer, and this implied longer cycle times. These interactions decreased the integrity of the model. However, in

the sensitivity analysis of the weights, we determined that neither orbit transfer capability nor cycle time had a noticeable effect on the results. From this we concluded that the relationship between these two measures did not affect the model conclusions. As the next section shows, the first place alternative was also highly insensitive to performance fluctuations in both of these measures, which further confirmed the acceptability of this model's results.

4.9.2 Performance Sensitivity Analysis

This section addresses the impacts of variations in alternative performance. We wanted to know what it would take for the best alternative, number 22, to lose its top ranking. The difference in overall value between 22 and 16 is 0.2567 units. The following table delineates the change that must occur in each performance measure to drop Alternative 22 from first place.

Table 4.9-2. Performance Sensitivity Analysis Results

Measure	Current Score	Score to Lose 1st Place
Mean Repair Time	7 days	56 days
Cycle Time	0.29 years	2.5 years
Upgrade Frequency	4 missions	1 mission
ORU Capacity	150 kg	105 kg
RDT&E Cost	$136.9 mil	$275 mil
3 or 6 Planes	6 planes	3 planes
Multi-Usability	High	Medium
Transfer Capability	Phase & Transfer	0

For the most part, these results indicated a robust number one ranking for Alternative 22. The ORU Capacity, 3 or 6 Planes and Multi-Usability measures were the most sensitive. A 30% drop in ORU capacity yielded a new top alternative. This was a parameter that resulted from a combination of customer requirements and the level of flexibility in the design. The GPS JPO controls the latter influence but not the former. Further study into potential customer requirements could address the former concern. The 3 or 6 Plane measure reflected the possibility that GPS may modify its architecture from the current six-plane configuration to a

three-plane configuration. This research may contribute to that debate if the JPO chooses on-orbit servicing as part of their future constellation architecture. In that case, it would be important to consider this sensitivity of the top alternative. Alternative 22 was sensitive to a one-level change on the Multi-Usability measure. If technology development or cost limited the servicer to medium multi-usability, the decision for best alternative would change. Feasibility studies into the expected success of various servicer technologies could reduce the uncertainty of performance in this measure. We assessed the above sensitivities one measure at a time. It is more likely that variability would appear in several parameters. This analysis shows that some statistically significant sensitivities exist, and it will be important for users of this research to further explore these impacts as requirements and technology evolve.

4.9.3 Sensitivity of Mean Time to Repair

The Mean Time to Repair measure warranted some individual attention. We used simulation to understand the impacts of repair alternatives. We built the model in AweSim and used IIA failure distribution data to check that the model worked properly. We discovered that the primary drivers of mean repair time were the repair policies and whether the constellation had one servicer or one servicer per plane. With this all in place, several exchanges occurred between GPS and us in an effort to acquire IIF reliability data. However, we were never able to acquire the data we needed. Thus, the data in the value model comes from the IIA output. We used sensitivity analysis to assess the impact on our results if this approximate data varied significantly from the true data.

The method we used to assess the potential impact of variability in the data was to make uniform changes to the performance numbers. Increasing the mean time to repair scores uniformly by a factor of ten caused slight changes in the rankings, but the top three alternatives

remained in the top three. Decreasing the scores by a factor of ten had a similarly minor effect, and the top three alternatives still held the top three positions. Thus, it appears our estimation of IIF mean time to repair data based on the IIA data adequately represents the information for our purposes. Alternatives that stretch the limits of our analysis, however, may warrant new estimation of this parameter.

V Conclusions

5.1 Putting the Alternatives in Perspective

The following table shows the top three circled alternatives from Figure 4.8-1. These alternatives offer the best value at their cost, and, thus, dominate the alternatives below and to the right of them. It would be from these alternatives that the decision-maker would select the final on-orbit servicing candidate. The table below contains the design parameters and costs of these alternatives in decreasing order of overall value.

Table 5.1-1. Parameters of Top Three Boundary Alternatives

	Alternative		
Parameter	22	16	13
# GPS Planes	6	6	6
ORU Capacity (kg)	150	300	150
Servicer Capability	High	Medium	High
# Service Times	4	4	4
RS Design Life	15 years	15 years	15 years
RS Mass Total (kg)	715	387	639
RDT&E Cost	$136.9 Mil	$100.6 Mil	$136.9 Mil
Avg. Mission Cost	$338 Mil	$317 Mil	$290 Mil
	Alternative		
Parameter	21	20	18
# GPS Planes	3	6	3
ORU Capacity (kg)	150	50	50
Servicer Capability	High	Low	Medium
# Service Times	4	4	4
RS Design Life	15 years	15 years	15 years
RS Mass Total (kg)	1021	183	412
RDT&E Cost	$147.4 Mil	$100.6 Mil	$81.1 Mil
Avg. Mission Cost	$229 Mil	$123 Mil	$113 Mil

These six alternatives exhibit a broad range of possible scores in most areas with a few notable exceptions. They all conduct four servicing missions and have design lives of fifteen years, and they all come from architectures B and D. The B and D architectures

both have a servicer in each plane. Architecture B only has upgrade capability, and architecture D has upgrade and quick repair capability. The price tags on these alternatives range from $113 million to $338 million, and a comparison with the current alternative offers some perspective on the magnitude of those costs. If GPS decided to upgrade a constellation using current methods, it would have to launch 24 new satellites. At approximately $30 million per Block IIF satellite and $73 million per launch vehicle for the hardware alone, it would cost GPS $2.472 billion to perform one upgrade through constellation replacement. This is ten times the average mission costs of the first three alternatives and four times their costs over the design life of the constellation. This baseline method of launching a new constellation scores a fourth place overall performance of 8.225 in the value model. We added this GPS alternative to the following figure to emphasize these tradeoffs. The lone marker on the right is the full constellation replacement alternative.

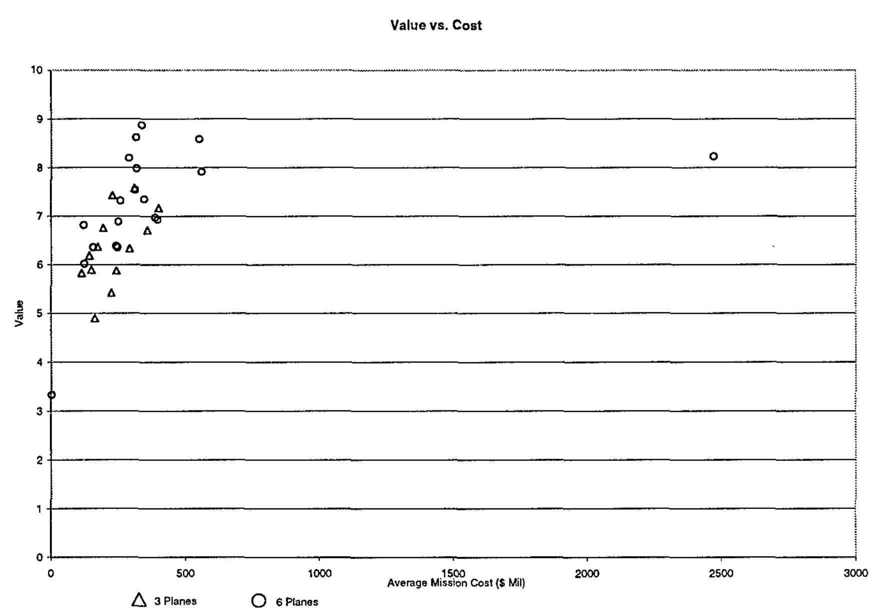


Figure 5.1-1. Comparison of Alternatives with Full Constellation Replacement

5.2 Process Summary

In addition to the specific analysis of on-orbit servicing alternatives, our sponsor was interested in the process through which we conducted the analysis. The following graphic depicts the approximate proportions of time that it took to accomplish our analysis from start to finish. These proportions would apply to any complex and technical decision situation.

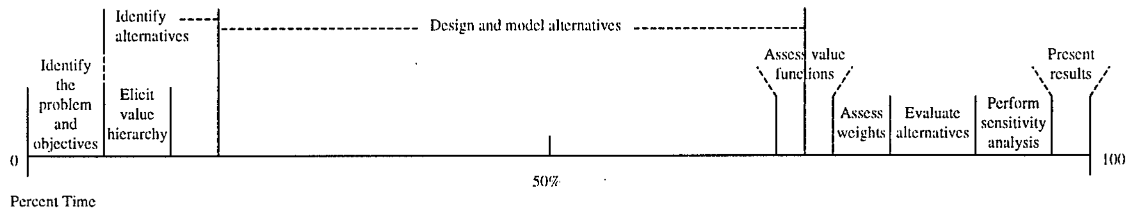


Figure 5.2-1. Process Time Allocation

Allowing for the appropriate amount of time and effort at each stage of the process will have a significant positive effect on the outcome of any future research. Frequent involvement of all stakeholders is critical to ensure cooperation and comprehension in the latter phases of the process. An understanding of the interdependencies of the methodology and the general timeline can greatly facilitate proper allocation of manpower and other resources. In the end, all of these aspects come together to build a complete picture that aids the decision-maker in his or her decision-making process.

5.3 Scope of Variations Analyzed

A summary of the variations in servicing alternatives can give an appropriate scope to our analysis. The top-level parameters we varied were the following:

- Eight different overall servicing architectures
    - Three different design lives for the robotic servicers
    - Three types of employment strategies

- Two different constellation plane sizes (3 or 6 orbital planes)
- 3 ORU capabilities
- 21 different Logistics and Transportation System concepts consisting of:
  - Four different launch vehicles
  - Three different upper stage systems
  - Three different dispenser capacities (one, three, or six payloads per launch vehicle)
  - Three different inclinations, and three different altitudes for low earth orbit parking orbits
- Three different robotic servicer concepts

5.4 Impact of Our Results

Our results – the analysis of on-orbit servicing alternatives and the process that guided the analysis – have potentially far reaching effects in the satellite community. GPS recognizes the need to explore evolving technologies that can increase constellation flexibility. They need the ability to deploy capabilities faster, and they would like the ability to market their satellites as platforms for customers outside the GPS JPO. Our study has shown that on-orbit servicing can deploy new capabilities in a rapid manner with reasonable cost. On-orbit servicing of the GPS constellation would give the U.S. the ability to quickly deploy global coverage space capabilities. In addition, on-orbit servicing deconflicts the drive to lower cost through longer satellite design lives from the need to respond quickly to changing requirements. In essence, GPS would evolve from a navigation satellite to a multiuse global platform.

To accomplish the goals of deploying capabilities faster, GPS will evaluate a wide variety of alternatives. On-orbit servicing is a category of those alternatives, and our results offer an analysis of thirty on-orbit architectures. At least as important as the specific results of this analysis was the process and framework for evaluating alternatives. GPS has been part of our process from the beginning and is in an excellent position to facilitate further work in a larger forum. Howard Wishner gave us continuous support throughout this effort. He has drafted a proposal to draw other satellite program managers and representatives into a discussion on the future of satellite operations.

GPS is not alone in this quest for better satellite management. The cancellation of the Flight Telerobotic Servicer did not negate the need for on-orbit servicing of the space station. With the initial deployment of the International Space Station, the Special Purpose Dexterous Manipulator is under development as an integral part of the assembly and maintenance of the station. The designers of Space Based Laser (SBL) have identified on-orbit servicing as an enabling technology for refueling of an operational system (Knutson, 1999). Just as SDI saw the benefits of on-orbit servicing in the 1980's, so SBL sees the benefits of servicing now. In November of 1997, the Modular On-orbit Servicing (MOS) Integrated Product Team (IPT) came into existence and is under the leadership of Dr. Rich Madison of AFRL-Kirtland. We have attended MOS IPT meetings and have actively participated in the development of their requirements document.

5.5 Influential Control of GPS

This thesis examined alternatives for the current constellation configuration of six planes, and it examined three-plane configuration alternatives. GPS is in the process of

determining their best configuration for the future, and they agreed that our research should evaluate both possibilities. The results, then, are a combination of two sets of data – one group for six planes and one group for three planes. We kept the data together to facilitate comparing the groups to each other and to the Baseline alternative. Statistical analysis revealed that, on average, the six-plane alternatives outperformed the three-plane alternatives. However, this research is not likely to be a driver in the three-plane versus six-plane decision. Once the JPO makes that decision, it will be possible to focus on the appropriate on-orbit servicing alternatives.

Prior to choosing an alternative category, GPS will more specifically define its constellation management requirements. When these are in place, and if on-orbit servicing is the category the JPO chooses, the requirements will help narrow the list of viable alternatives. GPS would determine which servicing features – repair or upgrade or both – to incorporate. A significant driver of possible choices is the level of satellite redesign that GPS will undergo. We based our analysis on a broad range of overall needs and a correspondingly flexible perspective towards potential satellite redesign.

5.6 Enabling Technologies

The topic of enabling technologies could be a study unto itself. In fact, the main purpose of AFRL's Modular On-Orbit Servicing Integrated Product Team is to identify the crucial technologies in this area. However, with the results of this study, we are able to make some general recommendations. The top three alternatives used the dispenser concept extensively. By having a parking orbit at a different altitude than the destination, it would be possible to deliver payloads to multiple orbital planes from one launch vehicle. This is critical to developing a low cost method of on-orbit servicing for a large

constellation. Fortunately, using a dispenser in LEO is becoming common place, and there should not be a significant leap in technology to incorporate upper stages on the payloads.

None of the three robotic servicing concepts were dominant within the best alternatives. In fact, robotic servicers were not the biggest contributor to the overall cost of the servicing system (Appendix Z). So are there any important technological considerations for on-orbit robots? One characteristic that maximized benefit while limiting cost was the ability to have long operational lives for the servicers. This was important not because of the production cost of the robotic servicer, but because of the cost of transporting the servicer to the operational orbit.

While none of the top three alternatives used an exotic robotic servicer propulsion system, propulsion technologies did play a critical factor in the Logistics and Transportation System. By using a solar thermal upper stage, we were able to use a much smaller launch vehicle, which dramatically minimized overall system cost. As one can see in Appendix Z, the launch costs account for over 50% of the recurring mission costs. Therefore, solar thermal and ion propulsion need to receive further research as operational upper stages.

5.7 Areas for Further Study

5.7.1 Identify Customer Requirements

Several objectives for a new constellation management methodology are likely to come from customers of the constellation. It is important to identify the customers who might benefit from an ability to put payloads into GPS orbit and customers interested in

global coverage. Researching their potential requirements can guide the decision-making process and focus alternative evaluation.

5.7.2 Feasibility Studies.

It may be that the enabling technologies for alternatives already exist. The research of this thesis intentionally focused on technologies that already exist or are in development. It will be necessary to investigate the progress of these technologies to better refine the timeline to test and field the alternative of choice. If an enabling technology is beyond the current thinking of researchers, it will be necessary to conduct feasibility studies for the enabling technologies.

5.7.3 Concepts for Further Analysis

Due to the breadth of the on-orbit servicing field, this thesis did not cover every concept available. One concept that could be used in more applications is the use of piggybacking payloads. With launch costs being a large portion of the overall system costs, a "free ride" to orbit has many benefits. The main drawback is this opportunity is very program specific, since many programs do not have the excess launch capacity. Another concept that could be analyzed is the use of electric propulsion for the precessing depository orbit architecture. This investigation would involve significant orbital dynamics analysis, but could provide a very beneficial alternative.

5.7.4 Value Hierarchy

As we mentioned in Section 4.9.1.1, there are some flaws in the value hierarchy from a theoretical standpoint. To improve the soundness of the results, it will be necessary to modify the hierarchy to eliminate dependencies among the measures.

5.7.5 Multivariate Sensitivity Analysis

The sensitivity analysis we performed in this research effort was one-dimensional in nature. We looked at the effects of varying one weight at a time and keeping the others proportionate to their original values. We also varied measure scores for the top alternative and determined the point of decision change. We did not attempt to assess the effects of varying two or more inputs simultaneously. The model was fairly robust to the univariate sensitivities we examined. However, future analysis may benefit from multivariate testing.

5.7.6 Cost – Benefit Tradeoff Analysis

Future research into this topic should include a systematic way to quantify the tradeoff between overall value and cost for each alternative. To be theoretically sound, any such quantification must consider the design of the value model and the methodology behind the assessed costs.

5.7.7 Simulation

The purpose of simulation in this thesis was to assess the impact of repair on each architecture's performance. The simulation could extend to include the benefits and interactions of upgrade and repair missions. Modeling both functions together would add fidelity to the simulation. Fuel and mass calculations could be a part of the simulation. The interactions between the servicer, the ORU needs for each repair mission, and the depot could be incorporated. Many other improvements could become part of the simulation.

5.7.8 Use of Statistics

The use of statistics to perform the means comparisons was based on the assumption that our alternatives were random samples of the solution space. Our choice of alternatives, however, was not entirely random. We selected alternatives that represented the full range of alternatives. Future analysis may benefit from using design of experiments and response surface methodology when selecting alternatives to evaluate.

5.7.9 Expanding the Application of This Process

This thesis provided both answers and needed tools for analyzing the benefits of on-orbit servicing. The concepts and processes outlined in our study of GPS could apply to other satellite programs. Having refined the analysis methodologies for specific application to satellite management alternatives, future users need only apply this process. This will save both time and effort.

5.8 Conclusion – Looking to the Future

The GPS JPO is in a position as an experienced, successful, and forward thinking satellite program to champion support for a new satellite management paradigm. This thesis defined and explained a thorough process for evaluating constellation architecture alternatives for the GPS program. This process can extend to evaluate alternatives for other satellite programs and for a composite group of programs in a cooperative forum. The satellite community could benefit greatly from a change in their methods, and the program that leads the way stands to benefit the most through its ability to guide the changes.

Satellites have become integral to the functioning of our society. They impact us in many ways from the morning weather and news to the navigation systems of mass transit to national security. Satellite program managers with an eye to the future know that they must find a way to keep up with the rapidly evolving demands on their satellite systems. This necessity becomes more apparent to the Department of Defense as more foreign militaries operate in space. As the U.S. continues to respond to threats around the world, military space systems offer the continuous, global coverage capabilities that are instrumental in achieving our objectives. However, to maintain a leadership role in space technology development, our military space systems must be responsive to their changing requirements. Flexibility becomes more of a challenge for larger satellite constellations such as GPS with 24 satellites or Iridium with 66 satellites. On-orbit servicing is a promising candidate to achieve this flexibility. It offers the ability to put new hardware on existing satellites and repair failed satellites. It could do this in a fraction of the time and cost it would take to design, build, and launch a new satellite system. It would allow the trend to reduce programs costs through longer satellite lives to continue, while providing a cost effective method of keeping the a satellite system's capabilities up to date.

Management with on-orbit servicing offers unique benefits most satellite programs do not have. Whether the U.S. military will go forward with this method is uncertain. What is certain is the growing need for a new satellite management paradigm. Programs such as GPS and SBL are actively investigating new solutions. Technology that exists now or is in development may hold the keys for managers to more efficiently maintain the currency of their satellite systems.

Appendix A-1: Development of Spreadsheet #1 – Insertion into LEO calculations

The following appendix describes the actual calculations derived for spreadsheet #1. Spreadsheet #1 performed the orbital dynamic, propulsion, and cost calculations to determine an entire LTS system using a LEO parking orbit. Refer to appendix A-2 or A-3 for examples.

Column 1: Altitude and inclination of parking orbit. Our objective is to make an equal comparison of different launch vehicles. However, launch vehicle manufactures list the performance with different LEO definitions, thus we used a spreadsheet with all the necessary orbits. In reality the parking orbit would be chosen to optimize launch vehicle performance and other mission objectives.

Col. 2: The semi-major axis is used to calculate required changes in orbital velocity (Delta V's) for the transfer burns. The transfer orbit has a perigee of the LEO parking orbit altitude, and apogee at the GPS orbital altitude.

Col. 3: Delta Omega Dot is the difference in the rate of precession of the longitude of ascending node between the LEO parking orbit and GPS's orbit.

Col. 4: Time to cover the GPS constellation is the time it requires for the canisters to precess to all the different orbits. The 6 orbital planes are 150 degrees apart (30 degrees between planes). Thus col. 4 is calculated by dividing 150 by Col. 3. For a 3-plane constellation the planes are 120 degrees apart (60 degrees between planes) and would have 20% decrease in time.

Col. 5: Impulsive Delta V at perigee is the change in velocity (km / s) for the perigee burn of a Hohmann transfer.

Col. 6: Impulsive Delta V at apogee is the change in velocity to complete the Hohmann transfer and perform any inclination change also. Inclination changes require less Delta V (and propellant) at apogee verses perigee.

Col. 7-10: The burnout mass ratio is the ratio of required initial mass (M_o) over the final mass (M_f). The larger the number the less the final payload. The mass ratio is found by manipulating the rocket equation (Wiesel, 1997:195) to get:

$$M_o / M_f = exp\ (v / V_e) \qquad \text{Equation 16}$$

where v is the velocity change (col. 5 and/or 6) in meters / second (m / s), and V_e is the effective exit velocity in m/s. We calculate V_e by multiplying the specific Impulse (Isp) by gravity (9.8 m / sec^2). To find the Isp for solids, we used an average value from Thiokol's Star family and Lockheed Martin's TOS motor (Wilson, 1994: 271, 287,8).

Table A-1. Star and TOS Motor I_{sp} Values

Star 27	288 seconds
Star 30BP	292 seconds
Star 37XFP	290 seconds
TOS motor	294 seconds
Value used in thesis	290 seconds

To find the Isp for liquid rockets, an average value for N_2O_4/MMH thrusters in the 100 – 500 Newton Thrust class (Larson and Wertz, 1992: 657) was applicable. The Air Force Research Laboratory's (AFRL) Solar Orbit Transfer Vehicle (SOTV) provided a representative Isp for solar thermal rockets of approximately 800 (Dornheim, 1998: 76).

Col. 7 & 8: Since a solid rocket motor cannot be used for multiple burns, it requires two motors for the two burns, and so the mass ratios are for the 1st and 2nd burns respectively.

Col. 9: The liquid rocket engine can be used for multiple burns. Thus, the same rocket engine performed both burns by combining the Delta V's and finding the mass ratio from before the perigee burn to after the apogee burn.

Col. 10: Solar thermal propulsion will requiring many burns but will execute them similar to a Hohmann transfer and is calculated like Col. 9. However, since it is low thrust the burn is not exactly at perigee and a kinematic inefficiency occurs. Solar thermal propulsion requires 7% more Delta V based on AFRL's SOTV (13,780 fps. versus 14,760 fps. [Dornheim, 1998: 77]).

Col. 11: Ion propulsion performs a spiraling transfer orbit over a long period of time. For more accurate results, different analysis than impulsive burn type calculations would have to be performed. However, based on a similar orbital transfer example in Spaceflight Dynamics (92) the kinematic inefficiency between impulsive and low thrust propulsion systems was 20%. Thus, ion propulsion performance was calculated using the above mentioned rocket equation, but the Delta V was 120% of the impulsive burn Delta V requirements.

Col. 12: Mass into LEO is an input variable (see LV section below).

Col. 13-16: Payload mass (Mp) is found by:

$$Mp = Mo * (Mf / Mo - Ms / Mo) \qquad \text{Equation 17}$$

Mo is initial mass which is the LV's mass to LEO minus the dispenser mass:

$$Mo = Mo - Mo * (\text{dispenser percentage}) \qquad \text{Equation 18}$$

We found the dispenser percentage by contacting Boeing (see section 4.4.3.7).

Mf / Mo is the inverse of the mass ratio from column 7-10.

Ms / Mo is the structural ratio. While this ratio can be defined different ways, we used the definition found in Space Mission Analysis and Design (SMAD) where Mo is defined by the total weight (Larson and Wertz, 1992: 669). This definition of the ratio enabled the use of the above equation. We chose Ms / Mo for the solid rocket motors to be 5%, determined by SMAD (658) and verified by analyzing the PAM-D which is:

189 / (2180 (fuel) + 1861 (payload)) = 5% Equation 19

(Wilson, 1994: 288)

We chose Ms / Mo for liquid rocket engines to be 7%, determined by SMAD (660). We verified this was a good approximation by comparing it to the IABS (Integrated Apogee Boost Subsystem) rocket for the Defense Satellite Communication System (DSCS) program.

227 / (1479 + 1180) = 8.5% Equation 20

(Wilson, 1994: 255)

Ms / Mo for solar thermal was 24% based on AFRL's SOTV.

3,600 / (8,000 + 6,700) = 24.5% Equation 21

Since GPS will be either a 3 or 6 orbital plane constellation, the most logical re-supply mission would be to 1, 3, or 6 planes. We grouped the boosters so that we had small, medium, and large categories of canisters. Each canister will go to one of GPS's orbital planes. Since the canister is mostly just a mechanical structure, we chose the structural ratio of 8% (Larson and Wertz, 1992: 321).

Appendix A-2: Spreadsheet #1 – Costs Are Averaged Over 8 Launches

#1: Insertion into LEO parking orbit

GPS orbit					
Semi-maj axis =	26600 km	inclination	=	0.98 rad =	56 deg
Omega dot =	-0.04 deg/day	velocity	=	3.87 km/sec	

LEO orbit	
altitude =	300 km
semi-maj. axis =	6678 km
	7.7 km/s

Transfer Orbit		
semi-maj. Axis =	16639 km	
omega dot =	-0.19 deg/day	for 56 deg inc.
omega dot =	0.26 "	for 28 deg

Notes for the below spreadsheet

3rd "	This is the difference in rates between GPS and the transfer orbit		*note	solid uses 2 stages / liquid & solar thermal use 1
5th "	Perigee burn from parking orbit to transfer orbit			thus final payload mass uses multi-stage rocket eq.
6th "	The burn to circularize the transfer orbit at apogee into a GPS orbit		*note	Mass to tran. Orbit is an input variable
7 & 8th	The solid motor (SMAD 677 (TOS) & Janes(Star)) has an Is	290	*note	final payload mass = M0*(Mf/Mo-ms/mo)
9th "	The liquid motor (N2O4 & MMH) has an Isp of	305		mo = MTO-MTO*(dispenser ratio)
10th "	The solar thermal (AWST 3/30/98) has an Isp(sec) of	800		s.r. = structural ration (ms/mo) (% is in next block -->)
11th "	The ion engine (DS #1 web page) has an Isp of	3100		dispenser ratio is dispenser mass/total mass

Ratios: Ms/Mo		
rocket	solid	0.05
	liquid	0.07
	sol th	0.245
disp	6 can	0.08
	3 can	0.07
	1	N/A
canister =		0.08

											EELV med (3 OTC's)					EELV M+ (5,4) (3 OTC's)					Delta II (1 target orbit)						
altitude&inc of parking	semi-maj. Axis of trans. orbit	delta omega dot (deg/day)	time to cover (days)	Implusive Delta V perigee	Implusive Delta V apogee	Mass ratio (solid (perigee))	Mass ratio (solid (apogee))	Mass ratio (liquid)	Mass ratio (solar thermal)	Mass ratio (ion)	Mass(kg) into LEO (kg)	payload mass to trans (solid)	Final payload mass (solid)	Final payload mass (liquid)	Final p/d mass (solar thermal)	Mass into Tran orbit (kg)	payload mass to trans (solid)	Final payload mass (solid)	Final payload mass (liquid)	Final p/d mass (solar thermal)	Mass into Tran orbit (kg)	payload mass to trans (solid)	Final payload mass (solid)	Final payload mass (liquid)	Final p/d mass (solar thermal)	Final p/d mass (ion)	Mass
28 deg =	0.489 rad.																										
185	16,582	7.96	18.6	2.145	2.07	2.13	2.07	4.09	1.71	1.15		0	0	0	0		0	0	0	0	5,092	2,139	927	888	1,728	4,178	
300	16,640	7.49	19.8	2.043	2.06	2.05	2.06	3.94	1.69	1.14		0	0	0	0		0	0	0	0		0	0	0	0	0	
460	16,720	6.90	21.5	1.905	2.05	1.95	2.06	3.75	1.66	1.14		0	0	0	0		0	0	0	0		0	0	0	0	0	
37 deg =	0.646 rad.																					1881	930	<--test case with GPS IIA masses			
185	16,582	7.20	20.6	2.145	1.76	2.13	1.86	3.69	1.64	1.14		0	0	0	0		0	0	0	0		0	0	0	0	0	3,975
300	16,640	6.78	21.9	2.043	1.75	2.05	1.85	3.55	1.62	1.13		0	0	0	0		0	0	0	0		0	0	0	0	0	
370	16,675	6.53	22.7	1.982	1.74	2.01	1.84	3.47	1.61	1.13		0	0	0	0		0	0	0	0		0	0	0	0	0	3,665
45 deg =	0.785 rad.																										
185	16,582	6.38	23.2	2.145	1.55	2.13	1.73	3.44	1.60	1.13		0	0	0	0		0	0	0	0		0	0	0	0	0	3,780
250	16,615	6.16	24.0	2.087	1.54	2.08	1.72	3.37	1.59	1.13	7,940	3,174	1,686	1,675	2,839	12,300	4,917	2,611	2,595	4,397		0	0	0	0	0	
370	16,675	5.78	25.6	1.982	1.53	2.01	1.71	3.24	1.56	1.12		0	0	0	0		0	0	0	0		0	0	0	0	0	3,480
56 deg =	0.977 rad.																										

	EELV med solid	EELV med liquid	EELV med solar th.	EELV M+ solid	EELV M+ liquid	EELV M+ solar th.	Delta II solid	Delta II liquid	Delta II solar th.	Delta II ion
mass of upper stage per each OTC =	2,085	2,088	1,700	3,230	3,235	7,903	4,165	4,204	3,364	
Total impulse (kg*s) / OTC unit =	6.E+05	6.E+05	6.E+05	9.E+05	9.E+05	9.E+05	1.E+06	1.E+06	1.E+06	

For determining sizing of the ORU's canister in each orbit (For example we say 42 degrees 250 km perigee is the optimal transfer orb

Launch Costs (99$)	
EELV Med. =	7.3E+07
EELV M+ (5,2). =	9.5E+07
Delta II=	4.9E+07
Kister K-1 =	3.3E+07
Taurus XL=	2.5E+07
Upper stage cost factors	
inflation factor =	1.249
Solid motor RD&TE factor =	0.1
Solid motor prod. Factor =	0.34
Liquid Eng. RD&TE factor =	0.8
Solar ther. RD&TE factor =	1.3
Solar ther. Prod. Factor =	0.85
Canister Cost Factors	
Canister RDT&E factor =	0.5
Dispenser Cost Factors	
fixed first unit cost ($92 K) =	3103
structural RD&TE factor =	0.2
Production scaling factor =	0.5

OTC (canister & up. stage) & Dispenser (Dis.) costs
learning curve slope = 0.9
b = 0.848
\# of missions = 8

	EELV med			EELV M+			Delta II			
OTC mass ------->	562	558	946	870	865	1,466	927	888	1,728	4,178
ORU mass ---------->	517	514	870	801	796	1,348	853	817	1,590	
	Medium Canisters						Big Canisters			
Up. St. Cost ($K) for RDT&E =	3,762	8,931	14,513	5,794	13,835	22,482	7,454	17,979	29,216	
Cost ($K) for 1st Upper stage Unit =	6,042	3,721	3,163	8,398	5,765	4,900	10,086	7,491	6,368	
Canister cost ($K) for RDT&E =	4,851	4,838	6,165	5,923	5,906	7,678	6,105	5,981	8,371	
Cost ($K) for 1st Unit canister =	1,274	1,269	1,788	1,694	1,687	2,377	1,765	1,716	2,646	
Average cost / OTC ($K) per mission (1, 3 or 6 units/mission) =	4,514	3,079	3,054	6,226	4,597	4,489	8,639	6,712	6,571	
Dispenser cost ($K) for RDT & E =	3,710	3,710	3,710	4,789	4,789	4,789				
Cost ($K) for 1st Unit dispenser =	5,205	5,205	5,205	6,281	6,281	6,281				
Average cost /dispenser per mission (1 unit / mission) =	3,795	3,795	3,795	4,579	4,579	4,579				
Total RDT&E cost for each alternative =	12,323	17,479	24,388	16,507	24,530	34,949	13,560	23,960	37,587	
Total cost ($K) (one mission) for OTC(s) and dispenser =	17,335	13,031	12,958	23,255	18,370	18,046	8,639	6,713	6,571	
For R.S. RDT&E for upper stage & dis =	7,472	12,641	18,223	10,583	18,624	27,271	7,454	17,979	29,216	
cost for average mission =	13,512	9,222	7,593	18,174	13,308	10,915	6,874	4,996	3,925	

For determining cost/kg. of ORU for each alternative (using launch vehicle, OTC, & dispenser costs)

EELV med			EELV M+			Delta II			
47,084	47,368	27,959	39,554	39,793	23,488	57,452	59,967	30,818	11,727

*note: I calculated ion propulsion only for a R.S. with an ion engine, this applies to the 1 target orbit scenerios
Also the ion engine is part of the final payload, thus I used a S.R. for solid motors just for the extra tankage
*note I added 20% to the required delta V for an ion engine based on a geosynchronous example in Wiesel p92
Kinematic ineff. Of ion engine = *0.2*

Kinematic Ineff. Of Sol. Ther = *0.07*

	Kistler K-1(1 orbit)						Delta II (3 OTC's)					Taurus XL (1 orbit)						EELV med (6 OTC's)				
)	s into Tran orbit (kg)	payload mass to trans (solid)	Final payload mass (solid)	Final payload mass (liquid)	Final p/d mass (solar thermal)	Final p/d mass (ion)	Mass into Tran orbit (kg)	payload mass to trans (solid)	Final payload mass (solid)	Final payload mass (liquid)	Final p/d mass (solar thermal)	Mass into Tran orbit (kg)	payload mass to trans (solid)	Final payload mass (solid)	Final payload mass (liquid)	Final p/d mass (solar thermal)	Final p/d mass (ion)	Mass(kg) into LEO (kg)	payload mass to trans (solid)	Final payload mass (solid)	Final payload mass (liquid)	Final p/d mass (solar thermal)
	0	0	0	0		5,092	1,990	862	826	1,607		0	0	0	0	0		0	0	0	0	
	0	0	0	0			0	0	0	0		0	0	0	0	0		0	0	0	0	
	0	0	0	0			0	0	0	0	1,400	646	282	275	503	1,159		0	0	0	0	
s																						
	1,670	816	799	1,443	3,297		0	0	0	0		0	0	0	0	0		0	0	0	0	
	0	0	0	0	0		0	0	0	0		0	0	0	0	0		0	0	0	0	
	1,642	808	799	1,382	3,059		0	0	0	0		0	0	0	0	0		0	0	0	0	
	1,588	841	833	1,433	3,158		0	0	0	0		0	0	0	0	0		0	0	0	0	
	0	0	0	0	0		0	0	0	0		0	0	0	0	0	7940	3,140	1,668	1,657	2,808	
	1,559	833	832	1,372	2,926		0	0	0	0		0	0	0	0	0		0	0	0	0	
		2,939	2,947	2,347				1,410	1,422	1,162			1,118	1,125	897				1,045	1,047	855	
		8.E+05	8.E+05	8.E+05				4.E+05	4.E+05	4.E+05			3.E+05	3.E+05	3.E+05				3.E+05	3.E+05	3.E+05	
it)																						
		841	833	1,433	3,158			287	275	536			282	275	503	1,159			278	276	468	
		773	766	1,318				264	253	493			260	253	463				256	254	431	
											Small Canisters											
		5,279	12,604	20,482				2,564	6,082	9,883			2,046	4,811	7,817				1,917	4,478	7,277	
		7,848	5,252	4,464				4,624	2,534	2,154			3,912	2,004	1,704				3,728	1,866	1,586	
		5,826	5,801	7,588				3,706	3,649	4,752			3,681	3,648	4,624				3,661	3,653	4,487	
		1,656	1,646	2,342				824	802	1,236			815	801	1,186				807	803	1,132	
		6,928	5,029	4,962				3,361	2,058	2,091			3,446	2,045	2,106				2,518	1,482	1,509	
								488	488	488									3,710	3,710	3,710	
								4,386	4,386	4,386									5,205	5,205	5,205	
								3,197	3,197	3,197									3,795	3,795	3,795	
		11,105	18,405	28,070				6,758	10,219	15,123			5,727	8,458	12,441				9,288	11,841	15,474	
		6,928	5,029	4,962				13,280	9,371	9,470			3,446	2,046	2,107				18,900	12,687	12,848	
		5,279	12,604	20,482				3,052	6,570	10,371			2,046	4,811	7,817				5,627	8,188	10,987	
		5,272	3,382	2,620				10,807	6,965	5,763			2,631	1,244	921				14,061	7,867	6,058	
		42,679	43,071	25,036				61,782	64,484	33,139			94,386	96,765	52,976				47,597	47,882	28,263	

Appendix A-3: Spreadsheet #1 – Costs Are Averaged Over 2 Launches

#1: Insertion into LEO parking orbit

GPS orbit					
Semi-maj axis =	26600 km	inclination	=	0.98 rad =	56 deg
Omega dot =	-0.04 deg/day	velocity	=	3.87 km/sec	

LEO orbit	
altitude =	300 km
semi-maj. axis =	6678 km
	7.7 km/s

Transfer Orbit		
semi-maj. Axis =	16639 km	
omega dot =	-0.19 deg/day	for 56 deg inc.
omega dot =	0.26 "	for 28 deg

Notes for the below spreadsheet				
3rd "	This is the difference in rates between GPS and the transfer orbit		*note	solid uses 2 stages / liquid & solar thermal use 1
5th "	Perigee burn from parking orbit to transfer orbit			thus final payload mass uses multi-stage rocket eq.
6th "	The burn to circularize the transfer orbit at apogee into a GPS orbit		*note	Mass to tran. Orbit is an input variable
7 & 8th	The solid motor (SMAD 677 (TOS) & Janes(Star)) has an Is	290	*note	final payload mass = M0*(Mf/Mo-ms/mo)
9th "	The liquid motor (N2O4 & MMH) has an Isp of	305		mo = MTO-MTO*(dispenser ratio)
10th "	The solar thermal (AWST 3/30/98) has an Isp(sec) of	800		s.r. = structural ration (ms/mo) (% is in next block -->)
11th "	The ion engine (DS #1 web page) has an Isp of	3100		dispenser ratio is dispenser mass/total mass

Ratios: Ms/Mo		
rocket	solid	0.05
	liquid	0.07
	sol th	0.245
disp	6 can	0.08
	3 can	0.07
	1	N/A
canister =		0.08

											EELV med (3 OTC's)					EELV M+ (5,4) (3 OTC's)					Delta II (1 target orbit)						
altitude&inc of parking	semi-maj. Axis of trans. orbit	delta omega dot (deg/day)	time to cover (days)	Implusive Delta V perigee	Implusive Delta V apogee	Mass ratio (solid (perigee))	Mass ratio (solid (apogee))	Mass ratio (liquid)	Mass ratio (solar thermal)	Mass ratio (ion)	Mass(kg) into LEO (kg)	payload mass to trans (solid)	Final payload mass (solid)	Final payload mass (liquid)	Final p/d mass (solar thermal)	Mass into Tran orbit (kg)	payload mass to trans (solid)	Final payload mass (solid)	Final payload mass (liquid)	Final p/d mass (solar thermal)	Mass into Tran orbit (kg)	payload mass to trans (solid)	Final payload mass (solid)	Final payload mass (liquid)	Final p/d mass (solar thermal)	Final p/d mass (ion)	Mass
28 deg = 0.489 rad.																											
185	16,582	7.96	18.6	2.145	2.07	2.13	2.07	4.09	1.71	1.15		0	0	0	0		0	0	0	0	5,092	2,139	927	888	1,728	4,178	
300	16,640	7.49	19.8	2.043	2.06	2.05	2.06	3.94	1.69	1.14		0	0	0	0		0	0	0	0		0	0	0	0	0	
460	16,720	6.90	21.5	1.905	2.05	1.95	2.06	3.75	1.66	1.14		0	0	0	0		0	0	0	0		0	0	0	0	0	
37 deg = 0.646 rad.																						1881	930	<--test case with GPS IIA masses			
185	16,582	7.20	20.6	2.145	1.76	2.13	1.86	3.69	1.64	1.14		0	0	0	0		0	0	0	0		0	0	0	0	0	3,975
300	16,640	6.78	21.9	2.043	1.75	2.05	1.85	3.55	1.62	1.13		0	0	0	0		0	0	0	0		0	0	0	0	0	
370	16,675	6.53	22.7	1.982	1.74	2.01	1.84	3.47	1.61	1.13		0	0	0	0		0	0	0	0		0	0	0	0	0	3,665
45 deg = 0.785 rad.																											
185	16,582	6.38	23.2	2.145	1.55	2.13	1.73	3.44	1.60	1.13		0	0	0	0		0	0	0	0		0	0	0	0	0	3,780
250	16,615	6.16	24.0	2.087	1.54	2.08	1.72	3.37	1.59	1.13	7,940	3,174	1,686	1,675	2,839	12,300	4,917	2,611	2,595	4,397		0	0	0	0	0	
370	16,675	5.78	25.6	1.982	1.53	2.01	1.71	3.24	1.56	1.12		0	0	0	0		0	0	0	0		0	0	0	0	0	3,480
56 deg = 0.977 rad.																											

	EELV med			EELV M+			Delta II			
mass of upper stage per each OTC =	2,085	2,088	1,700	3,230	3,235	7,903	4,165	4,204	3,364	
Total impulse (kg*s) / OTC unit =	6.E+05	6.E+05	6.E+05	9.E+05	9.E+05	9.E+05	1.E+06	1.E+06	1.E+06	

For determining sizing of the ORU's canister in each orbit (For example we say 42 degrees 250 km perigee is the optimal transfer orb

Launch Costs (99$)	
EELV Med. =	7.3E+07
EELV M+ (5,2). =	9.5E+07
Delta II=	4.9E+07
Kister K-1 =	3.3E+07
Taurus XL=	2.5E+07
Upper stage cost factors	
inflation factor =	1.249
Solid motor RD&TE factor =	0.1
Solid motor prod. Factor =	0.34
Liquid Eng. RD&TE factor =	0.8
Solar ther. RD&TE factor =	1.3
Solar ther. Prod. Factor =	0.85
Canister Cost Factors	
Canister RDT&E factor =	0.5
Dispenser Cost Factors	
fixed first unit cost ($92 K) =	3103
structural RD&TE factor =	0.2
Production scaling factor =	0.5

OTC (canister & up. stage) & Dispenser (Dis.) costs
learning curve slope = 0.9
b = 0.848
\# of missions = 2

	EELV med			EELV M+			Delta II			
OTC mass ------->	562	558	946	870	865	1,466	927	888	1,728	4,178
ORU mass ---------->	517	514	870	801	796	1,348	853	817	1,590	
	Medium Canisters						**Big Canisters**			
Up. St. Cost ($K) for RDT&E =	3,762	8,931	14,513	5,794	13,835	22,482	7,454	17,979	29,216	
Cost ($K) for 1st Upper stage Unit =	6,042	3,721	3,163	8,398	5,765	4,900	10,086	7,491	6,368	
Canister cost ($K) for RDT&E =	4,851	4,838	6,165	5,923	5,906	7,678	6,105	5,981	8,371	
Cost ($K) for 1st Unit canister =	1,274	1,269	1,788	1,694	1,687	2,377	1,765	1,716	2,646	
Average cost / OTC ($K) per mission (1, 3 or 6 units/mission) =	5,572	3,801	3,771	7,686	5,675	5,542	10,665	8,287	8,112	
Dispenser cost ($K) for RDT & E =	3,710	3,710	3,710	4,789	4,789	4,789				
Cost ($K) for 1st Unit dispenser =	5,205	5,205	5,205	6,281	6,281	6,281				
Average cost /dispenser per mission (1 unit / mission) =	4,685	4,685	4,685	5,653	5,653	5,653				
Total RDT&E cost for each alternative =	12,323	17,479	24,388	16,507	24,530	34,949	13,560	23,960	37,587	
Total cost ($K) (one mission) for OTC(s) and dispenser =	21,402	16,087	15,998	28,710	22,678	22,278	10,666	8,287	8,112	
For R.S. RDT&E for upper stage & dis =	7,472	12,641	18,223	10,583	18,624	27,271	7,454	17,979	29,216	
cost for average mission =	17,578	12,279	10,633	23,629	17,617	15,148	8,900	6,570	5,466	
For determining cost/kg. of ORU for each alternative (using launch vehicle, OTC, & dispenser costs)										
	47,087	47,370	27,960	39,557	39,795	23,489	57,454	59,969	30,819	11,727

*note: I calculated ion propulsion only for a R.S. with an ion engine, this applies to the 1 target orbit scenerios
Also the ion engine is part of the final payload, thus I used a S.R. for solid motors just for the extra tankage

*note I added 20% to the required delta V for an ion engine based on a geosynchronous example in Wiesel p92
Kinematic ineff. Of ion engine = *0.2*

Kinematic Ineff. Of Sol. Ther = *0.07*

	Kistler K-1(1 orbit)						Delta II (3 OTC's)					Taurus XL (1 orbit)						EELV med (6 OTC's)				
)	s into Tran orbit (kg)	payload mass to trans (solid)	Final payload mass (solid)	Final payload mass (liquid)	Final p/d mass (solar thermal)	Final p/d mass (ion)	Mass into Tran orbit (kg)	payload mass to trans (solid)	Final payload mass (solid)	Final payload mass (liquid)	Final p/d mass (solar thermal)	Mass into Tran orbit (kg)	payload mass to trans (solid)	Final payload mass (solid)	Final payload mass (liquid)	Final p/d mass (solar thermal)	Final p/d mass (ion)	Mass(kg) into LEO (kg)	payload mass to trans (solid)	Final payload mass (solid)	Final payload mass (liquid)	Final p/d mass (solar thermal)
	0	0	0	0		5,092	1,990	862	826	1,607		0	0	0	0	0		0	0	0	0	
	0	0	0	0			0	0	0	0		0	0	0	0	0		0	0	0	0	
	0	0	0	0			0	0	0	0	1,400	646	282	275	503	1,159		0	0	0	0	
s																						
	1,670	816	799	1,443	3,297		0	0	0	0		0	0	0	0	0		0	0	0	0	
	0	0	0	0	0		0	0	0	0		0	0	0	0	0		0	0	0	0	
	1,642	808	799	1,382	3,059		0	0	0	0		0	0	0	0	0		0	0	0	0	
	1,588	841	833	1,433	3,158		0	0	0	0		0	0	0	0	0		0	0	0	0	
	0	0	0	0	0		0	0	0	0		0	0	0	0	0	7940	3,140	1,668	1,657	2,808	
	1,559	833	832	1,372	2,926		0	0	0	0		0	0	0	0	0		0	0	0	0	
		2,939	2,947	2,347				1,410	1,422	1,162			1,118	1,125	897				1,045	1,047	855	
		8.E+05	8.E+05	8.E+05				4.E+05	4.E+05	4.E+05			3.E+05	3.E+05	3.E+05				3.E+05	3.E+05	3.E+05	
it)																						
		841	833	1,433	3,158			287	275	536			282	275	503	1,159			278	276	468	
		773	766	1,318				264	253	493			260	253	463				256	254	431	
												Small Canisters										
		5,279	12,604	20,482				2,564	6,082	9,883			2,046	4,811	7,817				1,917	4,478	7,277	
		7,848	5,252	4,464				4,624	2,534	2,154			3,912	2,004	1,704				3,728	1,866	1,586	
		5,826	5,801	7,588				3,706	3,649	4,752			3,681	3,648	4,624				3,661	3,653	4,487	
		1,656	1,646	2,342				824	802	1,236			815	801	1,186				807	803	1,132	
		8,553	6,208	6,126				4,149	2,540	2,582			4,254	2,525	2,600				3,108	1,830	1,863	
								488	488	488									3,710	3,710	3,710	
								4,386	4,386	4,386									5,205	5,205	5,205	
								3,947	3,947	3,947									4,685	4,685	4,685	
		11,105	18,405	28,070				6,758	10,219	15,123			5,727	8,458	12,441				9,288	11,841	15,474	
		8,553	6,208	6,126				16,395	11,569	11,692			4,254	2,525	2,601				23,334	15,663	15,862	
		5,279	12,604	20,482				3,052	6,570	10,371			2,046	4,811	7,817				5,627	8,188	10,987	
		6,897	4,562	3,783				13,922	9,163	7,985			3,440	1,724	1,415				18,494	10,843	9,072	
		42,681	43,072	25,037				61,786	64,487	33,141			94,389	96,767	52,977				47,600	47,884	28,264	

Appendix B: Spreadsheet #2 – Insertion into GPS Transfer Orbit

#2: Insertion into GPS transfer orbit

GPS orbit				
Semi-maj axis =	26600 km	inclination	0.9774 rad =	56 deg
Omega dot =	-0.0376 deg/day	velocity	: 3.8710 km/sec	

LEO orbit		
altitude	300	km
s.m. axis	6678	km

Transfer Orbit		
s.m. axis	16639 km	
omega dot	-0.19 d/day	for 56 deg incline
omega dot	0.26	for 28 deg. Incl.

Notes for the below spreadsheet			
1st column	185 km was the given for launch vehicle performance, however		
	Parking orbit will depend on air drag effects	*note	An electric Transfer Orbit covered under S.S. 3
3rd "	*This is the delta between GPS and the transfer orbit*	*note	*Mass to tran. Orbit is an input variable*
5th "	Circularize transfer orbit at apogee into GPS orbit	*note	final payload mass = M0*(Mf/Mo-ms/mo)
6th "	The solid motor has and Isp of 290 sec (SMAD 677 & Janes)		mo = MTO-MTO*(dispenser ratio)
7th "	The liquid motor has an Isp of 305 sec (N2O4 & MMH)		s.r. = structural ration (ms/mo) (% is in next block -->)
8th "	The solar thermal has an Isp of 800 sec (AWST 3/30/98)		dispenser ratio is dispenser mass/total mass

kinematic ineff of Sol. T=		
mass % deductions		
rm	solid	0.05
	liquid	0.07
	sol th	0.245
disp	6 can.	0.2
	3 can.	0.1
	1 can.	N/A

												EELV Medium (3 can)				EELV M+(5,4) (3 canisters)			
altitude (km)	semi-maj. Axis of tran. Orb.	delta omega dot (deg/day)	time to cover (days)	Implusive Delta V	Mass ratio (solid)	Mass ratio (liquid)	Mass ratio (solar thermal)	Mass into Tran orbit (kg)	Final payload mass (solid)	Final payload mass (liquid)	Final p/d mass (solar thermal)	Mass into Tran orbit (kg)	Final payload mass (solid)	Final payload mass (liquid)	Final p/d mass (solar thermal)	Mass into Tran orbit (kg)	Final payload mass (solid)	Final payload mass (	Final p/d
28 deg =	0.48869	radians																	
185	16582	-0.311	663.5	2.07	2.069	1.996	1.301	4,420	1,724	1,714	2,082	6,800	2,652	2,638	3,203	2,090	906	901	1,094
300	16639.5	-0.307	674.6	2.06	2.063	1.991	1.300		0	0	0		0	0	0		0	0	0
400	16689.5	-0.304	684.5	2.05	2.058	1.986	1.299		0	0	0		0	0	0		0	0	0
42 deg =	0.73304	radians																	
200	*16589.5*	*-0.261*	849.9	1.62	1.766	1.718	1.229		0	0	0		0	0	0		0	0	0
400	16689.5	-0.256	876.8	1.59	1.752	1.704	1.225		0	0	0		0	0	0		0	0	0
56 deg =	0.97738	radians																	
200	16589.5	-0.196	1340.4	1.43	1.656	1.615	1.201		0	0	0		0	0	0		0	0	0
400	16689.5	-0.192	1391.2	1.40	1.639	1.600	1.196		0	0	0		0	0	0		0	0	0

Launch Costs ($)

Delta II =	*5.5E+07*
EELV Med. =	*7.3E+07*
EELV M +(5,2)	*9.5E+07*
Kister K-1 =	*1.7E+07*
Taurus =	*2.5E+07*

Canister Structural Ratio
0.08

For determining sizing of the ORU's canister in each orbit (For example we say 42 degrees 2

canister mass -->	575	571	694	884	879	1068	906	901	1094
ORU mass -->	529	526	638	813	809	982	833	829	1006
	Medium Canisters			**Big Canisters**					

For determining cost/kg. of ORU for each launch vehicle

46,024	46,281	38,113	38,931	39,148	32,240	65,999	66,368	54,655

ation

0.07

Delta II (1 can.) Mass into Tran orbit (kg)	Final payload mass (solid)	Final payload mass (liquid)	Final p/d mass (solar thermal)	EELV med (6 canisters) Mass into Tran orbit (kg)	Final payload mass (solid)	Final payload mass (liquid)	Final p/d mass (solar thermal)	Delta II (3 canisters) Mass into Tran. Orbit (kg)	Final payload mass (solid)	Final payload mass (liquid)	Final p/d mass (solar thermal)	Taurus XL (1 orbit) Mass into Tran. Orbit (kg)	Final payload mass (solid)	Piggyback (Med) Mass into Tran. Orbit (kg)	Final payload mass (solid)	P/B (Med +)
4,420	1,533	1,524	1,851	2,090	815	811	984	695	301	300	364		0		0	
	0	0	0		0	0	0		0	0	0		0		0	
	0	0	0		0	0	0		0	0	0		0		0	
	0	0	0		0	0	0		0	0	0		0		0	
	0	0	0		0	0	0		0	0	0		0		0	
	0	0	0		0	0	0		0	0	0	704	390	3084	1,708	
	0	0	0		0	0	0		0	0	0		0		0	

;0 km perigee is the optimal transfer orbit)

	255	254	308 #		272	270	328		301	300	364		390		1708
	235	234	284 #		250	249	302		277	276	335		359		1572
	Small Canisters												Piggyback		

cost is that above normal IIF launch cost

	51,777	52,066	42,877		73,333	73,742	60,728		88,411	88,904	73,215		0		13,998

Appendix C-1: Development of Spreadsheet #3, page 1 – Phasing Within an Orbital Plane

The following appendix describes the calculations used in page 1 of spreadsheet #3. This spreadsheet was used to generate a generic table of mass ratios base on a set of different phasing times. Refer to appendix C-2 for an example.

Column 1: I chose a range of values that represent possible phasing times.

Col. 2: Period of phasing orbit is determined by the period of the original orbit (12 hours) plus the difference between the GPS and the phasing orbit periods. That difference is determined by the time for the target GPS S/V to get to the previous location of the RS (3 hrs) divided by the time allowed for total phasing (Col. 1).

Col. 3: Semi-major axis (a) is found by the following equation where P is the period (Col. 2) and μ is earth's gravitational constant.

$$a = [(P / 2\pi)^2 \mu]^{1/3} \qquad \text{Equation 22}$$

Col. 4: Energy of phasing orbit (E) is found by:

$$E = -\mu / 2a \qquad \text{Equation 23}$$

Col. 5: Velocity ($V_{phasing}$) needed to enter phasing orbit from GPS orbit is calculated by:

$$V = (2\,(E + \mu / r))^{1/2} \qquad \text{Equation 24}$$

where r is the radius at the maneuver, which is the GPS orbital radius.

Col. 6: Total required change in velocity (Delta V) is the 2 maneuvers times the delta V needed in each maneuver $V_{phasing} - V_{GPS}$. The maneuvers are to enter the phasing orbit and return to the GPS orbit.

Col. 7: Mass ratio is the rocket equation arranged like it was in spreadsheet #1.

Col. 8: Percent of S/C that is fuel is the percentage found by the mass ratio (e.g. a mass ratio of 1.096 has a fuel percentage of 9.6%)

Col. 9 – 15: To get mass ratios for multiple burns, I use the equation:

$$\textit{Total mass ratio} = (\textit{mass ratio for one burn})^{\wedge\, \#\, burns} \qquad \text{Equation 25}$$

Appendix C-2: Spreadsheet #3, Page 1

#3 (page 1) Phasing within the GPS orbit

GPS orbit		
Semi-maj axis	*26600*	km
Omega dot =	-0.0376	deg/day
velocity	3.871	(km/sec)
inclination	*0.9774*	rad

Chemical propulsion		Isp =	*320* sec
Solar Thermal	(to get upper limit delta V)	Isp =	*800* sec
Thrust (N) =	*34.6*	% orbit burned =	*0.33*
mass of R.S. =	*1500*	% inefficient =	*0.07*
accel. (km/s^2)	2.E-05	delta V per burn =	0.329 k/s

Notes for the below spreadsheet

*note	I'm finding average phasing manuever	4th	energy of phasing orbit
	(see GPS S/V's are 90 degrees apart)	6th	Delta V for both burns
*note	I'm always intercepting GPS's behind R.S.	8-13th	These are general characteristics to help pick days
	(less delta V needed to change periods of larger orbit		for phasing. See next page for more specifics
2nd column	period of phasing orbit (hrs)		*note: Solar thermal won't visit each S/V more than
	(=12hrs+ time for GPS to get to rendev. Point/2*(days of phasing)		once because it's not
3rd	SMAD p136, eq. 6-11		a long term propulsion system

6 plane GPS Constellation

Days for phasing	per. Of phasing	semi-major axis (km)	energy of phas.	velocity (km/s)	Total delta v (km/s)	mass ratio liquid	% of S/C is fuel	visit each S/V once	visit each S/V twice	visit each S/V four times	mass ratio solar thermal	% of S/C is fuel	visit each S/V once
elliptical phasing orbit - 2 burns								3 mane	19 man.	47man.			3 man.
1	13.500	28,784	-6.9240	4.015	0.288	1.0963	9.63	1.318	5.739	75.344	1.0401	4.01	1.125
2	12.750	27,708	-7.1929	3.948	0.153	1.0501	5.01	1.158	2.531	9.948	1.0211	2.11	1.065
4	12.375	27,162	-7.3375	3.911	0.080	1.0257	2.57	1.079	1.621	3.302	1.0109	1.09	1.033
8	12.188	26,887	-7.4125	3.892	0.041	1.0132	1.32	1.040	1.284	1.855	1.0056	0.56	1.017
12	12.125	26,795	-7.4380	3.885	0.028	1.0090	0.90	1.027	1.186	1.524	1.0038	0.38	1.012
16	12.094	26,749	-7.4508	3.882	0.022	1.0069	0.69	1.021	1.139	1.381	1.0029	0.29	1.009
20	12.075	26,721	-7.4585	3.880	0.018	1.0056	0.56	1.017	1.112	1.301	1.0024	0.24	1.007
30	12.050	26,684	-7.4688	3.877	0.012	1.0039	0.39	1.012	1.077	1.201	1.0017	0.17	1.005

3 plane GPS Constellation

								7 mane.	19 mane.	47 mane.			7 man.
1	12.750	27,708	-7.1929	3.948	0.153	1.0501	5.01	1.408	2.531	9.948	1.0211	2.11	1.158
2	12.375	27,162	-7.3375	3.911	0.080	1.0257	2.57	1.195	1.621	3.302	1.0109	1.09	1.079
4	12.188	26,887	-7.4125	3.892	0.041	1.0132	1.32	1.096	1.284	1.855	1.0056	0.56	1.040
8	12.094	26,749	-7.4508	3.882	0.022	1.0069	0.69	1.049	1.139	1.381	1.0029	0.29	1.021
12	12.063	26,703	-7.4637	3.878	0.015	1.0048	0.48	1.034	1.094	1.250	1.0020	0.20	1.014
16	12.047	26,680	-7.4701	3.877	0.012	1.0037	0.37	1.026	1.073	1.189	1.0016	0.16	1.011
20	12.038	26,666	-7.4740	3.876	0.010	1.0031	0.31	1.022	1.060	1.154	1.0013	0.13	1.009

Appendix D-1: Development of Spreadsheet #3, Page 2 – Calculations for R.S. propulsion Within an Orbital Plane

One of the main differences between the first (appendix C-1) and second steps (here) is the addition of carrying ORUs. In step one, we calculated the propellant needed to phase the RS with no ORUs. However, ORUs can be a large portion of the mass and thus step two includes their masses in the calculation. Including ORUs presents a problem related to multiple maneuvers. Multiple maneuvers were easy to calculate in step one. The total mass ratio for all the maneuvers was the mass ratio for one maneuver taken to the power of the maneuvers. The reason is exponential powers of mass ratios take into account the compounding effects of propellant needs. However, since ORU masses are fixed, we cannot use the rocket equation from step one. Amortization calculations take into account both a percentage based on the last value and a given value needed for each payment. This correlates to our phasing problem shown in the graphical example below (here we have six servicing missions with the RS picking up ORUs from the canister three times):

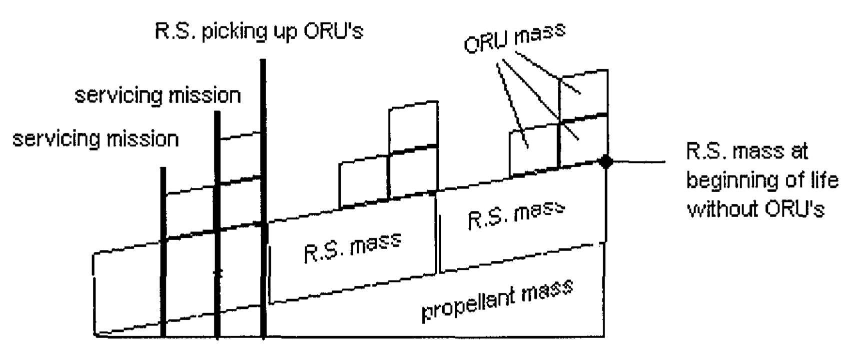


Figure D-1. Necessary Mass Proportions

Thus, to calculate the beginning mass of the RS, we find the future value function and subtract the ORU mass. However, we have two types of missions and two types of payment schedules for amortization.

Figure 4.4-9 is an example of the first type of mission used in Architectures A, E, and F.

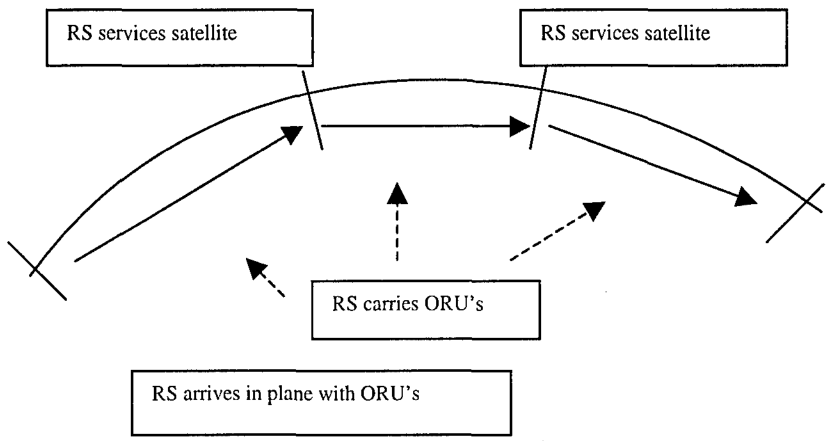


Figure D-2. A, E and F Mission Profile

This shows that the number of RS maneuvers is equal to the required servicing missions. Secondly, the RS's first maneuver is with ORUs. The future value (F.V.) function in Microsoft Excel refers to this as a "Type 1" payment method.

Below is a figure of the second type of mission, which was used in Architectures B, C, D, G, H.

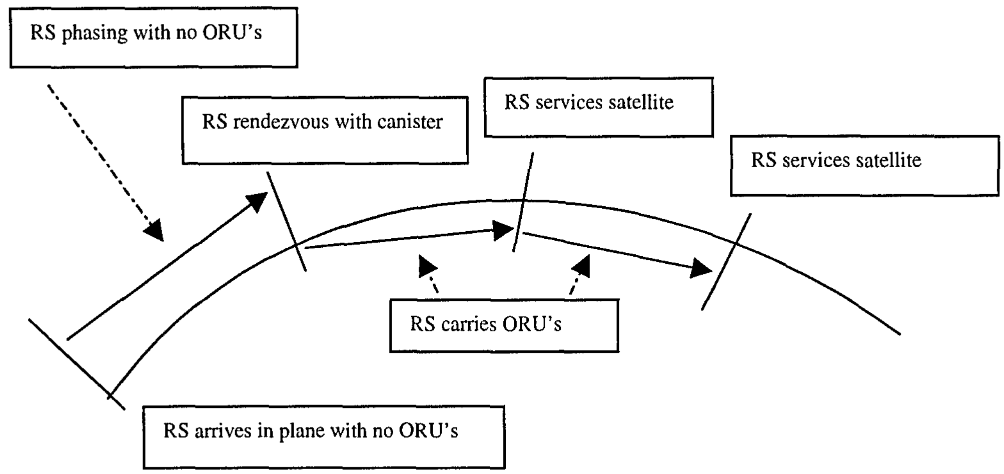


Figure D-3. B, C, D, G and H Mission Profile

The RS maneuvers one more time than the number of service missions. Also the first maneuver does not have ORUs. Therefore the future value function in Microsoft Excel refers to this a "Type 0" payment method.

The rest of the relation between an amortization calculation and phasing calculations is as follows:

Table D-1. Amortization Method of Tracking Mass

Interest rate = Percent of S/C that is fuel for one phasing maneuver
Total number of payments = number of maneuvers
Payment each period = ORU mass
Present Value = RS dry weight without ORUs

There are three independent input variables. The first variable is the mass of the RS excluding it's propulsion and ORUs (taken from output of RMS and RS bus analysis). The next variable is the number of GPS S/V's per plane (assume four S/V's for a 6-plane constellation and eight for a 3-plane constellation). The final variable is the mass of ORUs (taken from ORU analysis).

The first three rows show the input variables that vary between each alternative.

Row 1: We chose the number of visits to each GPS S/V in a plane based on the employment strategy for that alternative.

Row 2: Number of S/V's serviced per each servicing tour is the number of servicing missions between picking up more ORUs from a canister. Again, we chose this based on employment strategy.

Row 3: We picked the days for phasing from step 1 of this spreadsheet and based on the employment strategy.

Row 4: Mass of ORU pallet onboard the RS is calculated in this spreadsheet because it is a function of ORU mass. The mass of the pallet is calculated using a 6% structural ratio of total ORU mass carried at one time.

Row 5: The number of visits per plane is the number of GPS S/V's in that plane times the number of visits to each S/V.

Row 6: The number of servicing tours is the frequency of the RS rendezvousing with the OTC to acquire new ORUs. This variable is calculated by dividing the number of visits per plane by the number of S/V's serviced in each servicing tour. This variable determines how many iterations of the amortization calculation must be performed.

Row 7: The number of maneuvers is the total number of phasing orbits the RS must accomplish. This number includes visits to the GPS S/V's and to the OTC.

Row 8-14: The mass at the n^{th} servicing tour is the RS mass with the needed propellant and ORUs for those servicing missions.

Row 15: The mass at the 1^{st} servicing tour is the important output of this spreadsheet. This is the mass the Logistics and Transportation System will have to put in GPS orbit.

Row 16: The mass ratio is the initial mass of the RS in that orbit divided by the final mass. This ratio will be used in spreadsheet #4.

Row 17: The mass with initial ORUs is an optional output variable if the Air Force decides to launch the RS fully loaded with ORUs. This is used in the 10th Alternative of Architecture A.

Row 18: The minimum number of days for one servicing tour takes into account the time for all the phasing orbits and servicing times.

Appendix D-2: R.S. Propulsion Within an Orbital Plane – Low Capability, 50 kg Capacity, 6 Planes

#3 (page 2) (Mass calculations for phase changes)

mass of R.S. excluding propulsion (Mf-p)	138	kg	% for Mrocket/Mf-p for liquid rocket engine =	0.07	mass of R.S =	148 kg
Mass of ORU used at each servicing	50	kg	ion engine and power mass =	146	mass of R.S =	284 kg
Guess prop. mass for sol. th. (Al. H)=	3100	solar thermal rocket mass = 129	dry tot.=	439	mass of R.S =	577 kg

1st and 3rd rows are independent variables — days for servicing = 3

# of GPS S/V's in each plane =	4		**6 plane constellation**	(*note: change fuel % in mass ratios)			
Alternative	Alt. A	Alt. B	Alt. D	Alt. E	Alt. F	Alt. G	Alt. H
Employment Strategy	E.S. I	E.S. I	E.S. III	E.S. II	E.S. II	E.S. I	E.S. I
# of visits to each S/V in a plane	1	4	4	1	1	1	1
# of S/V's serviced / servicing tour		8	2	4	4	4	4
days for phasing	12	12	12	8	4	8	8
mass of ORU pallet	12	24	6	12	12	12	12
# visits per plane	4	16	16	4	4	4	4
Prop. & tank mass for free flyer	7	26	26	10	10	40	40
Prop. & tank for ff (150kg ORU)	13	53	53	20	20	79	79
# of servicing tours		2	8				
# manuevers	4	18	24	4	4	5	5
mass at last servicing tour			157				
mass at 7th servicing tour			160				
mass at 6th servicing tour			164				
mass at 5th servicing tour			167				
mass at 4th servicing tour		197	170				
mass at 3rd servicing tour		225	174				
mass at 2nd servicing tour		254	178				
mass at 1st servicing tour	170	286	180	319	190	319	605
mass ratio (Minitial/Mfinal)	1.15	1.94	1.22	1.12	1.29	1.12	1.05
mass with initial ORU's	370	686	280 (for 2 servicing missions)			519	805
minimum number of days for one servicing tour	60	132	42 *	44	28	52	52

*note: 12 days is average
for emergency it could be 6 days
and nonemergency 18 days

Appendix D-3: R.S. Propulsion Within an Orbital Plane – Medium Capability, 50 kg Capacity, 3 Planes

#3 (page 2) (Mass calculations for phase changes)

mass of R.S. excluding propulsion (Mf-p)	203 kg	% for Mrocket/Mf-p for liquid rocket engine =		0.07	mass of R.S =	217 kg
Mass of ORU used at each servicing	50 kg	ion engine and power mass =		146	mass of R.S =	349 kg
Guess prop. mass for sol. th. (Al. H)=	3100	solar thermal rocket mass =	129	dry tot.= 439	mass of R.S =	642 kg

1st and 3rd rows are independent variables — days for servicing = 3

# of GPS S/V's in each plane =		8	**3 plane constellation**		(*note: change fuel % in mass ratios)		
Alternative	Alt. A	Alt. B	Alt. D	Alt. E	Alt. F	Alt. G	Alt. H
Employment Strategy	E.S. I	E.S. I	E.S. III	E.S. II	E.S. II	E.S. I	E.S. I
# of visits to each S/V in a plane	1	4	4	1	1	1	1
# of S/V's serviced / servicing tour		8	2	8	8	8	8
days for phasing	12	12	12	8	4	8	8
mass of ORU pallet	24	24	6	24	24	24	24
# visits per plane	8	32	32	8	8	8	8
Prop. & tank mass for free flyer	13	53	53	5	5	40	40
Prop. & tank for ff (150kg ORU)	26	106	106	10	10	79	79
# of servicing tours		4	16				
# manuevers	8	36	48	8	8	9	9
mass at last servicing tour			228				
mass at 7th servicing tour			232				
mass at 6th servicing tour			237				
mass at 5th servicing tour			242				
mass at 4th servicing tour		272	246				
mass at 3rd servicing tour		305	251				
mass at 2nd servicing tour		341	256				
mass at 1st servicing tour	276	379	260	439	345	439	707
mass ratio (Minitial/Mfinal)	1.27	1.74	1.20	1.26	1.59	1.26	1.10
mass with initial ORU's	676	779	360 (for 2 servicing missions)			839	1,107
minimum number of days for one servicing tour	120	132	42 *	88	56	96	96

*note: 12 days is average
for emergency it could be 6 days
and nonemergency 18 days

Appendix D-4: R.S. Propulsion Within an Orbital Plane – Medium Capability, 150 kg Capacity, 3 Planes

#3 (page 2) (Mass calculations for phase changes)

mass of R.S. excluding propulsion (Mf-p)	263 kg	% for Mrocket/Mf-p for liquid rocket engine =		0.07	mass of R.S =	281 kg
Mass of ORU used at each servicing	150 kg	ion engine and power mass =		146	mass of R.S =	409 kg
Guess prop. mass for Alt. H (sol. Th.)=	0	solar thermal rocket mass =	190 dry tot.=	190	mass of R.S =	453 kg

1st and 3rd rows are independent variables — days for servicing = 3

3 plane constellation (*note: change fuel % in mass ratios)

# of GPS S/V's in each plane = 8							
Alternative	Alt. A	Alt. B	Alt. D	Alt. E	Alt. F	Alt. G	Alt. H
Employment Strategy	E.S. I	E.S. I	E.S. III	E.S. II	E.S. II	E.S. I	E.S. I
# of visits to each S/V in a plane	1	4	4	1	1	1	1
# of S/V's serviced / servicing tour		8	2	8	8	8	8
days for phasing	12	12	12	8	4	8	8
mass of ORU pallet	72	72	18	72	72	72	72
# visits per plane	8	32	32	8	8	8	8
Prop. & tank mass for free flyer	13	53	53	5	5	40	40
Prop. & tank for ff (150kg ORU)	26	106	106	10	10	79	79
# of servicing tours		4	16				
# manuevers	8	36	48	8	8	9	9
mass at last servicing tour			306				
mass at 7th servicing tour			313				
mass at 6th servicing tour			320				
mass at 5th servicing tour			327				
mass at 4th servicing tour		418	335				
mass at 3rd servicing tour		488	342				
mass at 2nd servicing tour		563	349				
mass at 1st servicing tour	429	643	355	657	581	608	580
mass ratio (Minitial/Mfinal)	1.53	2.28	1.26	1.61	2.06	1.49	1.28
mass with initial ORU's	1,629	1,843	655 (for 2 servicing missions)			1,808	1,780
minimum number of days for one servicing tour	120	132	42 *	88	56	96	96

*note: 12 days is average
for emergency it could be 6 days
and nonemergency 18 days

Appendix D-5: R.S. Propulsion Within an Orbital Plane – Medium Capability, 300 kg Capacity, 6 Planes

#3 (page 2) (Mass calculations for phase changes)

mass of R.S. excluding propulsion (Mf-p)	263 kg	% for Mrocket/Mf-p for liquid rocket engine =	0.07	mass of R.S =	281 kg
Mass of ORU used at each servicing	300 kg	ion engine and power mass =	146	mass of R.S =	409 kg
Guess prop. mass for Alt. H (sol. Th.)=	0	solar thermal rocket mass = 190 dry tot.=	190	mass of R.S =	453 kg

1st and 3rd rows are independent variables — days for servicing = 3

# of GPS S/V's in each plane = 4		6 plane constellation		(*note: change fuel % in mass ratios)			
Alternative	Alt. A	Alt. B	Alt. D	Alt. E	Alt. F	Alt. G	Alt. H
Employment Strategy	E.S. I	E.S. I	E.S. III	E.S. II	E.S. II	E.S. I	E.S. I
# of visits to each S/V in a plane	1	4	4	1	1	1	1
# of S/V's serviced / servicing tour		8	2	4	4	4	4
days for phasing	12	12	12	8	4	8	8
mass of ORU pallet	72	144	36	72	72	72	72
# visits per plane	4	16	16	4	4	4	4
Prop. & tank mass for free flyer	7	26	26	10	10	40	40
Prop. & tank for ff (150kg ORU)	13	53	53	20	20	79	79
# of servicing tours		2	8				
# manuevers	4	18	24	4	4	5	5
mass at last servicing tour			326				
mass at 7th servicing tour			334				
mass at 6th servicing tour			343				
mass at 5th servicing tour			352				
mass at 4th servicing tour		534	361				
mass at 3rd servicing tour		651	370				
mass at 2nd servicing tour		776	380				
mass at 1st servicing tour	394	911	387	594	470	547	554
mass ratio (Minitial/Mfinal)	1.40	3.24	1.38	1.45	1.67	1.34	1.22
mass with initial ORU's	1,594	3,311	987 (for 2 servicing missions)			1,747	1,754
minimum number of days for one servicing tour	60	132	42 *	44	28	52	52

*note: 12 days is average
for emergency it could be 6 days
and nonemergency 18 days

Appendix D-6: R.S. Propulsion Within an Orbital Plane – High Capability, 85 kg (Average) Capacity, 3 Planes

#3 (page 2) (Mass calculations for phase changes)

mass of R.S. excluding propulsion (Mf-p)	*494* kg	% for Mrocket/Mf-p for liquid rocket engine =	*0.07*	mass of R.S =	529 kg
Mass of ORU used at each servicing	*85* kg	ion engine and power mass =	*146*	mass of R.S =	640 kg
Guess prop. mass for Alt. H (sol. Th.)=	*0*	solar thermal rocket mass = *190* dry tot.=	190	mass of R.S =	684 kg

1st and 3rd rows are independent variables — days for servicing = *3*

# of GPS S/V's in each plane =		*8*	**3 plane constellation**		(*note: change fuel % in mass ratios)		
Alternative	Alt. A	Alt. B	Alt. D	Alt. E	Alt. F	Alt. G	Alt. H
Employment Strategy	E.S. I	E.S. I	E.S. III	E.S. II	E.S. II	E.S. I	E.S. I
# of visits to each S/V in a plane	*1*	*4*	*7*	*1*	*1*	*1*	*1*
# of S/V's serviced / servicing tour		*8*	*8*	8	8	*8*	*8*
days for phasing	*12*	*12*	*12*	*8*	*4*	*8*	*8*
mass of ORU pallet	40.8	40.8	40.8	40.8	40.8	40.8	40.8
# visits per plane	8	32	56	8	8	8	8
Prop. & tank mass for free flyer	13	53	92	5	5	40	40
Prop. & tank for ff (150kg ORU)	26	106	185	10	10	79	79
# of servicing tours		4	7				
# manuevers	8	36	63	8	8	9	9
mass at last servicing tour			634				
mass at 7th servicing tour			703				
mass at 6th servicing tour			777				
mass at 5th servicing tour			856				
mass at 4th servicing tour		634	942				
mass at 3rd servicing tour		703	1,034				
mass at 2nd servicing tour		777	1,132				
mass at 1st servicing tour	640	856	1,213	847	781	798	776
mass ratio (Minitial/Mfinal)	1.21	1.62	2.29	1.32	1.48	1.25	1.13
mass with initial ORU's	1,320	1,536	1,893 (for 2 servicing missions)			1,478	1,456
minimum number of days for one servicing tour	120	132	132 *	88	56	96	96

*note: 12 days is average
for emergency it could be 6 days
and nonemergency 18 days

Appendix D-7: R.S. Propulsion Within an Orbital Plane – High Capability, 300 kg Capacity, 3 Planes

#3 (page 2) (Mass calculations for phase changes)

mass of R.S. excluding propulsion (Mf-p)	*494* kg	% for Mrocket/Mf-p for liquid rocket engine =	*0.07*	mass of R.S =	529 kg
Mass of ORU used at each servicing	*300* kg	ion engine and power mass =	*146*	mass of R.S =	640 kg
Guess prop. mass for Alt. H (sol. Th.)=	*0*	solar thermal rocket mass = *190* dry tot.=	190	mass of R.S =	684 kg

1st and 3rd rows are independent variables — days for servicing = *3*

3 plane constellation (*note: change fuel % in mass ratios)

# of GPS S/V's in each plane = *8*							
Alternative	Alt. A	Alt. B	Alt. D	Alt. E	Alt. F	Alt. G	Alt. H
Employment Strategy	E.S. I	E.S. I	E.S. III	E.S. II	E.S. II	E.S. I	E.S. I
# of visits to each S/V in a plane	*1*	*4*	*7*	*1*	*1*	*1*	*1*
# of S/V's serviced / servicing tour		*8*	*8*	8	8	*8*	*8*
days for phasing	*12*	*12*	*12*	*8*	*4*	*8*	*8*
mass of ORU pallet	144	144	144	144	144	144	144
# visits per plane	8	32	56	8	8	8	8
Prop. & tank mass for free flyer	13	53	92	5	5	40	40
Prop. & tank for ff (150kg ORU)	26	106	185	10	10	79	79
# of servicing tours		4	7				
# manuevers	8	36	63	8	8	9	9
mass at last servicing tour			800				
mass at 7th servicing tour			936				
mass at 6th servicing tour			1,083				
mass at 5th servicing tour			1,240				
mass at 4th servicing tour		800	1,410				
mass at 3rd servicing tour		936	1,592				
mass at 2nd servicing tour		1,083	1,787				
mass at 1st servicing tour	822	1,240	1,947	1,067	1,120	1,018	928
mass ratio (Minitial/Mfinal)	1.56	2.35	3.68	1.67	2.12	1.59	1.36
mass with initial ORU's	3,222	3,640	4,347 (for 2 servicing missions)			3,418	3,328
minimum number of days for one servicing tour	120	132	132 *	88	56	96	96

*note: 12 days is average
for emergency it could be 6 days
and nonemergency 18 days

Appendix E-1: Development of Spreadsheet #4, page 1 – Ion propulsion for Architecture G

This spreadsheet calculates the propulsion system for an ion propulsion direct plane robotic servicer. Appendices E-2 and E-3 provide examples of these spreadsheets.

Again, finding the propellant mass and transfer times are dependent on each other. Transfer time depends on total mass of the RS and propellant mass depends on size of the dry RS, which depends on the size of the ion or solar thermal thruster. To find a good solution to this iterative problem, we chose the size of the propulsion unit (dry) to get certain acceleration, and thus transfer time, based on a certain sized robotic servicer.

For the ion propulsion system, we sized the propulsion unit to give .3 Newtons of thrust. Then we made an initial guess on the size of the RS and found the acceleration (row 7) based on Force = mass * acceleration. Then we calculated the change in the longitude of the ascending node angle (row 8) by:

$$\textit{Delta angle / orbit} = (4 * (\textit{radius of orbit})^2 * \textit{acceleration}) / \mu \qquad \text{Equation 26}$$

(Wiesel, 1997: 94, eq. 3.69)

The number of orbits (row 9) to change between orbital planes is calculate by:

$$\textit{\# orbit} = \textit{180 degrees} / (\textit{6 planes} * \textit{delta angle/orbit}) \qquad \text{Equation 27}$$

Since a GPS orbit has a 12-hour period the time of plane change is:

$$\textit{Time} = \textit{\# of orbits} / 2 \qquad \text{Equation 28}$$

The Total Delta Velocity will be needed later and is found by:

$$\textit{Total Delta V} = \textit{acceleration} * \textit{time} \qquad \text{Equation 29}$$

The total time to perform the plane changes depends on a 3 or 6-plane constellation

A 3-plane constellation requires 120 degrees of plane change, which corresponds to the time of 2nd, 3rd, 4th, and 5th plane changes. The 1st plane change is not included

because that is the 30 degrees that makes the difference between 120 and 150 degree total plane change. The 2nd and 4th plane change times are found by averaging the 1st and 3rd and the 3rd and the 5th.

A 6-plane constellation requires 150 degrees of total plane change, which corresponds to all 5-plane changes.

Total time for plane change and phasing (row 12) is the total time required to upgrade the GPS constellation. This is found by:

*Total time = total phasing time + (number of days between ORU pick up [spreadsheet 3, page 2])*number of planes* Equation 30

Notice spreadsheet 3 can only be set up for either a 3- or 6-plane constellation. This means only one of the total times in spreadsheet #4 can be right at one time. This particularity is the purpose for the warning statement in the middle of row 12.

Fuel needed (rows 13 – 16) finds the needed propellant mass for plane changes and altitude changes by the rocket equation

First, the rocket's Isp is converted to the effective exit velocity (Ve) by the multiplying it by standard gravity at sea level (9.8 m/s^2).

Next the mass ratio is found by the rocket equation

Mo / Mf = exp (delta V/Ve) Equation 31

We can find the total mass ratio (row 15) of both phasing and plane changes by:

$Total\ mass\ ratio = (mass\ ratio)_{phasing} * (mass\ ratio)_{plane\ change}$ Equation 32

We can find the mass ratio for the transfer from LEO to GPS orbit (row 16) by the same calculations of spreadsheet #1.

The illustrative mission scenario produces the numbers for evaluation of this alternative architecture, and also shows why chemical propulsion with low Isp is not attractive.

The RS masses are from spreadsheet #3.

The required total mass for each subsequent plane change maneuver is the RS mass in that orbit multiplied by the mass ratio for both a plane change maneuver and the phasing for one orbital plane. We presented each plane so the growth in required propellant mass could be shown.

In the chemical analysis we didn't show each plane but used the fifth power of the mass ratio to calculate the required propellant mass for all 5-plane changes and phasing maneuvers.

Appendix E-2: Ion Propulsion for Architecture G – Medium Capability, 150 kg Capacity, 6 Planes

#4 Alternative G: Low cost Upgrader (ion propulsion)

Below is row numbers

General Data

1	u(earth) =	398,601 km^3/s^2		GPS radius =	26,600 km
2	30 degrees between planes			velocity (GPS orbit) =	3.871 (km/sec)

Delta V between orbit plans using Impulsive Thrust (chemical)

3 deltaV=2vsin(angle/2) = 2.004 (km/s)

Total Delta V for 300 km LEO to GPS orbit (from Spread sheet #1)

4 deltaV = 4.103 (km/s)

Time needed to go between planes (Using Ion propulsion) (Wiesel p94)

5 (Thrust = .3 Newtons (Using 3 of Deepspace-1's ion engines - main requirement 7.5 kW power)

(*because of circular arguments, you need to manually put in mass in accel cell)

6	Last plane change:mass	426	kg.	1st pl. change:mass =	1408		3rd pl. ch	775
7	accel=	6.5E-07	Km/s^2	accel=	1.2E-07		accel=	2.8E-07
8	delta angle =	0.00463	rad/orbit	delta angle	0.00086		delta angle	0.00200
9	time	113.071	orbits	time	610.582		time	262.275
		56.535	days		305.291	days		131.138
10	delta V =	3.186	km/s	delta V =	3.186		delta V =	3.186
11	total 2 plane changes =	500	days	total 5 plane changes (rough) =				805 days
12	total plane and phasing	812	<- CHECK 3 vs. 6 .->	total 5 plane changes & phasing				1,117 days
		27	months					37 mon

Fuel needed (mass ratios)

	Ion propulsion		Chemical propulsion	
	Mass ratio for plane changes (ion prop.)		Mass ratio for plan change (chem. prop.)	
13	Ve = Isp*g0 (Isp=3100)	30380	Ve= Isp*g0 (Isp=350)	3430.000
14	mass ratio (mo/mf)	1.111	mass ratio (mo/mf)	1.794
15	mass ratio (with phasing)	1.348		2.253
	Mass ratio from LEO to GPS orbit (ion prop.)		Mass ratio from LEO to GPS orbit (chem)	
	(I assumed low thrust is 80% kinematically efficient as impulsive(ex p 92 Wiesel)		mass rat.	3.308
16	mass rat.	1.197	Mass ratio of Apogee burn (chem. prop.) GPS T.O is 45 degree with perigee 250 km	
			Mass ratio	1.567

ILLUSTRATIVE MISSION SCENARIO

(Electrical Propulsion) (for ORU mass of = 150)

17	Assume Spacecraft dry weight =		363.000 (kg)	Ion engine & power system mass=	146
18				RMS and R.S. bus mass =	217
			mass in GPS orbit	mass in LEO	
19	Service 2 planes	(1 plane ch	489	586	for 3 plane const.
20	Service 3 planes	(2 plane)	660	790	<-- 1 plane change
21	Service 4 planes	(3 plane)	889	1,065	
22	Service 5 planes	(4 plane)	1,199	1,435	<-- 2 plane changes
23	Service 6 planes	(5 plane)	1,616	1,935	

(Chemical Propulsion)

24	Assume Spacecraft dry weight =		247 (kg)		
			mass in GPS orbit	mass in LEO	mass in transfer orbit
25	Service 2 planes	(1 plane ch	444	1,468	695
26	Service 6 planes		32,320	106,901	50,636

Appendix E-3: Ion Propulsion for Architecture G – Medium Capability, 50 kg Capacity, 6 Planes

#4 Alternative G: Low cost Upgrader (ion propulsion)

Below is row numbers

General Data

1	u(earth) =	398,601	km^3/s^2	GPS radius =	26,600	km
2	30 degrees between planes			velocity (GPS orbit) =	3.871	(km/sec)

Delta V between orbit plans using Impulsive Thrust (chemical)

3 deltaV=2vsin(angle/2) = 2.004 (km/s)

Total Delta V for 300 km LEO to GPS orbit (from Spread sheet #1)

4 deltaV = 4.103 (km/s)

Time needed to go between planes (Using Ion propulsion) (Wiesel p94)

5 (Thrust = .3 Newtons (Using 3 of Deepspace-1's ion engines - main requirement 7.5 kW power)

(*because of circular arguments, you need to manually put in mass in accel cell)

6	Last plane change:mass	405	kg.	1st pl. change:mass =	925	3rd pl. ch	612
7	accel=	6.5E-07	Km/s^2	accel=	1.2E-07	accel=	2.8E-07
8	delta angle =	0.00463	rad/orbit	delta angle	0.00086	delta angle	0.00200
9	time	113.071	orbits	time	610.582	time	262.275
		56.535	days		305.291 days		131.138
10	delta V =	3.186	km/s	delta V =	3.186	delta V =	3.186

11	total 2 plane changes =	500	days	total 5 plane changes (rough) =	805	days
12	total plane and phasing	812	<- CHECK 3 vs. 6 .->	total 5 plane changes & phasing	1,117	days
		27	months		37	mon

Fuel needed (mass ratios)

	Ion propulsion		Chemical propulsion	
	Mass ratio for plane changes (ion prop.)		Mass ratio for plan change (chem. prop.)	
13	Ve = Isp*g0 (Isp=3100)	30380	Ve= Isp*g0 (Isp=350)	3430.000
14	mass ratio (mo/mf)	1.111	mass ratio (mo/mf)	1.794
15	mass ratio (with phasing)	1.230		1.990
	Mass ratio from LEO to GPS orbit (ion prop.)		Mass ratio from LEO to GPS orbit (chem)	
	(I assumed low thrust is 80% kinematically efficient as impulsive(ex p 92 Wiesel)		mass rat.	3.308
16	mass rat.	1.197	Mass ratio of Apogee burn (chem. prop.) GPS T.O is 45 degree with perigee 250 km	
			Mass ratio	1.567

ILLUSTRATIVE MISSION SCENARIO

(Electrical Propulsion)

				(for ORU mass of =	50)	
17	Assume Spacecraft dry weight =		363.000 (kg)	Ion engine & power system mass=	146	
18				RMS and R.S. bus mass =	217	
			mass in GPS orbit	mass in LEO		
19	Service 2 planes	(1 plane ch	446	534		for 3 plane const.
20	Service 3 planes	(2 plane)	549	657		<-- 1 plane change
21	Service 4 planes	(3 plane)	675	808		
22	Service 5 planes	(4 plane)	830	994		<-- 2 plane changes
23	Service 6 planes	(5 plane)	1,021	1,222		

(Chemical Propulsion)

24	Assume Spacecraft dry weight =		247 (kg)		
			mass in GPS orbit	mass in LEO	mass in transfer orbit
25	Service 2 planes	(1 plane ch	444	1,468	695
26	Service 6 planes		15,374	50,851	24,087

Appendix F-1: Development of Spreadsheet #4, page 2 – Solar Thermal Propulsion for Architecture H

The following appendix describes the design of the solar thermal propulsion system used for direct plane changes. Appendix F-2 is an example of this spreadsheet.

In the LTS analysis (col. 13-16, spreadsheet #1) I used mass ratios derived from SOTV because it was a similar mission (i.e. altitude change mission). However, the mission for RS propulsion is dramatically different. This can be seen in the requirement of 10+ km/s for 5 plane changes (row 4, page 2); whereas, the Delta V requirement for LEO to GPS transfer is 4.1 km/s (row 4, page 1). Thus using mass ratios from the SOTV would be inappropriate. Instead, we used actual thrust and mass numbers from SOTV and sized the propellant tank (see section 4.4.4.1).

The thrust levels of solar thermal make it a cross between high thrust chemical rockets and low thrust electric rockets. To analyze its effects on orbital maneuvers, we used the impulsive, chemical rocket equations, with two modifications to make those equations applicable to solar thermal rockets. Since solar thermal has a longer firing time than chemical, there is a kinematic inefficiency compared to impulsive type burns. This requires 7% more Delta V (LTS write-up, col. 10). With the much lower thrust than chemical propulsion, solar thermal requires many burns to accomplish the plane change. Thus, the spreadsheet calculates time for plane change like ion propulsion. However, unlike ion propulsion, solar thermal is analyzed by delta V / burn (row 5, page 2) versus change in angle (row 8, page 1).

The solar thermal propulsion alternative (page 2) is analyzed like ion propulsion (page 1) except for the following rows:

Row 1: Kinematic inefficiency of a solar thermal rocket burn profile compared to an impulsive type burn.

Row 2: Thrust is based off AFRL's SOTV (see mass ratio analysis)

Row 3: Length of burn was measured by percentage of orbit. With solar thermal propulsion being a new technology, there was no standard percentage. We chose to burn for 20% of the orbit.

Row 5: Delta V per orbit is calculated by multiplying the acceleration (row 4) times the time of the burn (calculated by percentage of orbit * 12 hrs.)

Row 6: Time (in orbits) to perform the orbit transfer was calculated by dividing the total Delta V for a plane change using solar thermal propulsion by the Delta V per orbit, and then rounded up. Total Delta V for solar thermal propulsion was found by multiplying an impulsive Delta V (row 3, page 1) by the kinematic inefficiency (row 1, page 2).

Row 21: Total propellant mass was found by subtracting the RS dry weight (spreadsheet #3, page 2) from the total weight (row 6). This measure is use to see how good the guessed propellant mass is in spreadsheet #3.

Appendix F-2: Solar Thermal Propulsion for Architecture H – Medium Capability, 150 kg Capacity, 6 Planes

#4 (Page 2) Alternative H:(solar thermal propulsion)

Below is row numbers

Like electric propulsion, solar thermal could do multiple plane changes
However, I calculate Time different than both electric and chemical
1 I calculate Delta V requirements from Impulsive burns with and inefficency factor being *0.07*
(Inefficency based on SOTV, AW&ST 3/30/98)

Time for plane change
2 (Thrust= 34.6 Newtons) (AWST 3/30/98) = *34.6*
3 (thruster burns % of the orbit) = *0.2*

Row	Last plane char	1053	kg.	1st pl. change:mass =	4260		3rd pl. ch	2118	
4	accel=	2.7E-05	Km/s^2	accel=	8.0E-06	Km/s^2	accel=	1.5E-05	Km/s^2
5	Delta V / orbit	0.23557	km/s	Delta V / o.	0.06910	km/s	Delta V / o.	0.12759	km/s
6	time	9.101	orbits	time	31.027	orbits	time	16.804	orbits
	(full orbits)	10.000		(full orbits)	32.000		(full orbits)	17.000	
		5.0	days		16.0	days		8.5	days
7	delta V =	2.144	km/s	delta V =	2.144	km/s	delta V =	2.144	km/s

Row						
8	2 (60deg/ch) plane change	33	days	total 5 plane changes (rough) =	49	days
9	total 2 plane + phasing=	345	<-check->	total 5 plane changes and phasing =	361	days
					12	mon

Mass ratios for solar thermal

Row	Mass ratio for plane change		Mass ratio from LEO to GPS	
10	Ve=Isp*g0 (Isp=800)	7840	Ve=Isp*g0 (Isp=800)	7840
11	Mass ratio	1.315	Mass ratio	2.110
12	mass ratio (with phasing)	1.418		

Mass ratio from GPS T.O into GPS final
(GPS T.O has a 45 degree inclination with 250 Km perigee.)

Row		
13	Ve=Isp*g0 (Isp=800)	7840
14	Mass ratio	1.234

(Solar Thermal Propulsion) (This is for a R.S. = 217 & ORU = 150)
15 R.S. weight in final plane = 871 (kg) (page 2 of spreadsheet 3)

Row		mass in GPS orbit	mass in LEO	mass in transfer orbit
16	Service 2 planes (1 plane cha.)	1,235	2,605	1,524
17	Service 3 planes	1,751	3,695	2,161
18	Service 4 planes	2,484	5,241	3,065
19	Service 5 planes	3,523	7,433	4,347
20	Service 6 planes	4,996	10,542	6,165
21	total propellant mass =	4,126	9,671	5,294

Appendix G-1: Development of Spreadsheet #5 (Processing Depository Propulsion)

This appendix describes the design of the propulsion unit used for the precessing depository architectures. Refer to appendix G-2 and G-3 for examples of this spreadsheet.

The calculations needed for the precessing depository orbit is exactly like the transfer orbit from LEO to GPS orbit. Spreadsheet #2 (Insertion into GPS transfer orbit) provided a basis for this spreadsheet. The only difference was how the propellant mass ratio was used (Col. 7 and following). To reduce redundancy, this section only explains the calculations which were different from spreadsheet #2.

The procedure for analyzing these alternatives was to calculate the wet RS masses for each leg of the servicing mission starting with the last leg first. Thus, we start with the RS dry mass for that alternative. Then, we calculate the propellant needed for the final leg of the mission, which is the RS without ORUs coming from the GPS orbit to the precessing orbit. Then add the propellant mass for the servicing missions, and so forth. The following paragraphs describe the actual calculations used in the spreadsheet.

Col. 7 & 14: RS mass before the return leg is the mass at the completion of the servicing missions. The return leg goes from the GPS orbit to the precessing depository orbit. This is calculated by multiplying the appropriate mass ratio (Col. 7 or 8) by the RS dry mass (an input variable).

Col. 8 & 15: RS mass at the start of the GPS orbit is the mass from the last column plus the ORUs and propellant to phase between the satellites. This is calculated by multiplying the last column times the mass ratio of the phasing orbit plus the total ORU mass for servicing the orbital plane.

Col. 9 & 16: The RS mass at the start of the tour is the mass in the depository orbit with all the propellant and ORUs for servicing the GPS S/V's in an orbital plane. This is calculated by multiplying the last column by the appropriate mass ratio. The mass ratios for liquid and solar thermal propulsion are in column 6 and 7 respectively.

Col. 10 & 17: Propellant needed for servicing one orbital plane is calculated by taking the last column and subtracting the RS dry mass.

Col. 11 & 18: Propellant + Tank mass for 6 trips is the mass the LTS system would have to place in the depository orbit. This is calculated by multiplying the last column by 6 and then adding structural mass ratio in the input column.

Col. 12 & 19: Propellant + Tank mass for 3 trips is calculated the same as the last column but only for 3 trips.

Appendix G-2: Precessing Depository Propulsion - Medium Capability, 150 kg Capacity for Upgrade and 20 kg Capacity for Repair, 6 Planes

#5: Propellant Cost for One On-orbit Depot (Alt. E & F)

GPS orbit					**LEO orbit**			**Transfer Orbit**		s.m. axis	16639 km
Semi-maj axis :	*26600* km	inclination	*0.98* rad		altitude	*300* km		omega dot	-0.19 d/day	for 56 deg inc.	
Omega dot	-0.04 deg/day	velocity	3.87 km/s		s.m. axis	6678 km		omega dot	0.26	for 28 deg. inc.	

Notes for the below spreadsheet		**Inputs** (either independat inputs or from spreadsheet #3)	
1st colu	185 km was the given for launch vehicle performance, however	Number of S/V's serviced in each plane =	4
	Parking orbit will depend on air drag effects	mass of repair ORU = *20* ORU total mass for one tour =	80
		ms/mf (2x phasing engine): 0.14 R.S. mass with liquid rocket engine :	247
3rd "	This is the delta between GPS and the transfer orbit	estimate tank size= *1300* Total R.S. (chem. Prop.) mass (dry) =	338
		estimate sol. th tank size= *1700* Total R.S. (sol. Thermal) mass (dry)=	577
5th "	Circularize transfer orbit at apogee into GPS orbit	Mass ratio (Mo/mf) for phasing mission (with ST) =	2.50
		Mass ratio (Mo/mf) for phasing mission (with liq.) =	1.45
13-14th	Propellant and tank is the mass of the resupply mission	Propellant Tank mass ratio (sol therm) = *0.1* Prop. Tank m. ratio (ch)	*0.07*

							Chemical propulsion						Solar Thermal Propulsion					
altitude (km)	semi-maj. Axis of tran. Orb.	delta omega dot (deg/day)	time to cover (days)	Implusive Delta V	Mass ratio (liquid)	Mass ratio (solar thermal)	R.S. mass before return leg	R.S. mass at the start of GPS	R.S. mass at start of tour	Propellant needed (1 trip)	Propellant + Tank (6 trips)	Propellant + Tank (3 trips)	R.S. mass before return leg	R.S. mass at the start of GPS	R.S. mass at start of tour	Propellant needed (1 trip)	Propellant + Tank (6 trips)	Propellant + Tank (3 trips)
28 deg	0.49	radians																
185	16582	-0.311	663	2.07	2.00	1.30	675	1,059	2,113	1,786	11,467	5,733	751	1,960	3,912	3,584	23,656	11,828
300	16640	-0.307	675	2.06	1.99	1.30	674	1,056	2,103	1,775	11,397	5,698	750	1,958	3,897	3,570	23,561	11,781
400	16690	-0.304	684	2.05	1.99	1.30	672	1,054	2,093	1,766	11,338	5,669	750	1,956	3,885	3,558	23,481	11,741
42 deg	0.73	radians																
200	16590	-0.261	850	1.62	1.72	1.23	581	922	1,584	1,257	8,070	4,035	709	1,855	3,187	2,859	18,871	9,435
400	16690	-0.256	877	1.59	1.70	1.23	577	916	1,560	1,233	7,916	3,958	707	1,850	3,152	2,825	18,643	9,321
56 deg	0.98	radians																
200	16590	-0.196	1340	1.43	1.62	1.20	547	872	1,409	1,081	6,942	3,471	693	1,814	2,930	2,603	17,178	8,589
400	16690	-0.192	1391	1.40	1.60	1.20	541	864	1,383	1,055	6,775	3,387	690	1,807	2,891	2,564	16,921	8,460

Cost (for 300km, 28 deg)		RT&E factor	*0.2*	tank mass=	746	373	2142	1071
inflation =	*1.249*	number of missior	*6*	RDT&E =	8,835	5,834	17,066	11,043
learning curve =	*0.9*	b=	0.8	1 mission=	4,823	3,074	9,577	6,103

Appendix G-3: Precessing Depository Propulsion - Low Capability, 50 kg Capacity for Upgrade and 20 kg Capacity for Repair, 6 Planes

#5: Propellant Cost for One On-orbit Depot (Alt. E & F)

GPS orbit				LEO orbit		Transfer Orbit		
Semi-maj axis :	*26600* km	inclination	*0.98* rad	altitude	*300* km	s.m. axis		16639 km
Omega dot	-0.04 deg/day	velocity	3.87 km/s	s.m. axis	6678 km	omega dot	-0.19 d/day	for 56 deg inc.
						omega dot	0.26	for 28 deg. inc.

Notes for the below spreadsheet	
1st colu	185 km was the given for launch vehicle performance, however Parking orbit will depend on air drag effects
3rd "	This is the delta between GPS and the transfer orbit
5th "	Circularize transfer orbit at apogee into GPS orbit
13-14th	Propellant and tank is the mass of the resupply mission

Inputs (either independat inputs or from spreadsheet #3)			
Number of S/V's serviced in each plane =			4
mass of repair ORU =	*20*	ORU total mass for one tour =	80
ms/mf (2x phasing engine):	0.14	R.S. mass with liquid rocket engine :	163
estimate tank size=	*1300*	Total R.S. (chem. Prop.) mass (dry) =	254
estimate sol. th tank size=	*1200*	Total R.S. (sol. Thermal) mass (dry)=	417
Mass ratio (Mo/mf) for phasing mission (with ST) =			1.28
Mass ratio (Mo/mf) for phasing mission (with liq.) =			1.28
Propellant Tank mass ratio (sol therm) =	*0.07*	Prop. Tank m. ratio (ch)	*0.07*

altitude (km)	semi-maj. Axis of tran. Orb.	delta omega dot (deg/day)	time to cover (days)	Implusive Delta V	Mass ratio (liquid)	Mass ratio (solar thermal)	Chemical propulsion: R.S. mass before return leg	R.S. mass at the start of GPS	R.S. mass at start of tour	Propellant needed (1 trip)	Propellant + Tank (6 trips)	Propellant + Tank (3 trips)	Solar Thermal Propulsion: R.S. mass before return leg	R.S. mass at the start of GPS	R.S. mass at start of tour	Propellant needed (1 trip)	Propellant + Tank (6 trips)	Propellant + Tank (3 trips)
28 deg	0.49	radians																
185	16582	-0.311	663	2.07	2.00	1.30	507	729	1,455	1,212	7,782	3,891	543	775	1,548	1,305	8,377	4,188
300	16640	-0.307	675	2.06	1.99	1.30	506	727	1,448	1,205	7,735	3,868	542	775	1,542	1,299	8,341	4,170
400	16690	-0.304	684	2.05	1.99	1.30	505	726	1,442	1,199	7,695	3,848	542	774	1,538	1,295	8,311	4,155
42 deg	0.73	radians																
200	16590	-0.261	850	1.62	1.72	1.23	436	639	1,097	854	5,482	2,741	513	737	1,266	1,023	6,565	3,282
400	16690	-0.256	877	1.59	1.70	1.23	433	634	1,081	838	5,377	2,689	511	735	1,252	1,009	6,478	3,239
56 deg	0.98	radians																
200	16590	-0.196	1340	1.43	1.62	1.20	410	605	978	735	4,716	2,358	501	722	1,165	922	5,922	2,961
400	16690	-0.192	1391	1.40	1.60	1.20	406	600	960	717	4,602	2,301	499	719	1,150	907	5,824	2,912

Cost (for 300km, 28 deg)								
inflation =	*1.249*	RT&E factor	*0.2*	tank mass=	506	253	545.7	272.8
learning curve =	*0.9*	number of missior	*6*	RDT&E =	6,990	4,666	7,313	4,870
		b=	0.8	1 mission=	3,749	2,389	3,937	2,509

Appendix H: Quick Lookup Tables

#6 Quick look up tables

ORU Transport Canister (OTC) size

		kgs.								
# tours			1		2		3		4	
#SV/tour			4	8	4	8	4	8	4	8
#S/V serviced			4	8	8	16	12	24	16	32
repair (average ORU)	------->	20		160	160	320	240	480	320	640
	.--->	60	240	480	480	960	720		960	
		100	400	800	800		1200			
upgrade (3 sizes)	.--->	150	600	1200	1200					
		250	1000							
	.--->	300	1200							

	up to
small =	250
med =	500
big =	800
XL =	1300

# of GPS S/V's in each plane = 8

Alternative	Alt. A	Alt. B	Alt. D	Alt. E	Alt. F	Alt. G	Alt. H
Employment Strategy	E.S. I	E.S. I	E.S. IV	E.S. II or II	E.S. II or III	E.S. I	E.S. I
# of visits to each S/V in a plane	1	1	2	1	1	1	1
				upgrade analysis			
# visits per plane	8	8	16	8	8	8	8
# of visits to each S/V in a plane			4				
# visits per plane			32				
# of visits to each S/V in a plane			6				
# visits per plane			48				

Calculations for Alt. D

		#/ S/V	mass	Mass / plane (with overhead)	
upgrade ORU mass:	150	4	600		
				3-plane	6-plane
repair ORU mass:	20	7	140	1288	616
average	85				

overhead for 3-plane = 0.15
overhead for 6-plane = 0.1

Calculations for Alt. E

		freq. / S/V	mass	depot	total depot
upgrade ORU mass:	150	4	600		
repair ORU mass:	20	7	140	3360	3528
average	85				

overhead for 1 depot = 0.05

Appendix I: NAFCOM Cost Sheet – Low Capability, 120 Day Design Life RS

Project: New Estimate
File Name: C:\NAFCOM96\Low_R.S._120day.ncm

NAFCOM 96 COST MODEL
WBS COST
1999 M$

Prepared By:
Revision:

	WBS Element	DDT&E	D&D	STH	Flight Unit	Production	Total	DDT&E FBS Factor	D&D FBS Factor	STH FBS Factor	FU FBS Factor
1	GRAND TOTAL	44.6			10.1	51.1	95.7	1.000			1.000
2	SYSTEM 1: Free Flying Servicer	44.6			10.1	51.1	95.7	1.000			1.000
3	HARDWARE TOTAL	27.6	18.6	9.0	6.9	35.2	62.8		1.000	1.000	1.000
4	R.S. Propulsion	1.7	1.0	0.7	0.6	2.8	4.5		1.000	1.000	1.000
5	Propulsion Unit (Chemical)	1.7	1.0	0.7	0.6	2.8	4.5		1.000	1.000	1.000
6	Payload	7.4	5.0	2.4	1.8	9.3	16.6		1.000	1.000	1.000
7	microsatellite	7.4	5.0	2.4	1.8	9.3	16.6		1.000	1.000	1.000
8	structure & mechanical	1.7	1.4	0.3	0.2	1.2	3.0		1.000	1.000	1.000
9	Avionics	4.8	3.0	1.8	1.4	7.0	11.8		1.000	1.000	1.000
10	Power system (batteries)	0.8	0.5	0.3	0.2	1.0	1.8		1.000	1.000	1.000
11	Arms & End effectors	0.0	0.0	0.0	0.0	0.0	0.0		1.000	1.000	1.000
12	Spacecraft Bus	18.5	12.6	5.9	4.6	23.2	41.7		1.000	1.000	1.000
13	Thermal control	1.0	0.8	0.2	0.1	0.7	1.7		1.000	1.000	1.000
14	Structures & Mechanisms	2.1	1.6	0.5	0.4	2.0	4.0		1.000	1.000	1.000
15	ADAC's & RCS	9.9	6.7	3.3	2.5	12.7	22.6		1.000	1.000	1.000
16	Data Management System	1.4	0.9	0.5	0.4	1.8	3.2		1.000	1.000	1.000
17	Communication System	0.6	0.3	0.3	0.2	1.2	1.9		1.000	1.000	1.000
18	Electrical Power System	3.5	2.3	1.2	0.9	4.7	8.2		1.000	1.000	1.000
19	SYSTEM INTEGRATION SUBTOTAL	17.0			3.1	15.9	32.9	1.000			1.000
20	IACO	2.0			0.9	4.3	6.4				
21	STO	0.9			N/A	N/A	0.9				
22	GSE										
23	Tooling	0.4			N/A	N/A	0.4				
24	M/E GSE	3.5			N/A	N/A	3.5				
25	SE&I	5.9			1.4	7.1	13.0				
26	PM	3.0			0.9	4.4	7.4				
27	LOOS	1.3			N/A	N/A	1.3				
28	Contingency	0.0			0.0	0.0	0.0				
29	Program Support	0.0			0.0	0.0	0.0				
30	Fee	0.0			0.0	0.0	0.0				
31											
32											
33	FUNCTIONAL RATES (FY96$)										
34	Engineering Labor Hourly Rate: $34.00										
35	Manufacturing Labor Hourly Rate: $28.00										

Project: New Estimate
File Name: C:\NAFCOM96\Low_R.S._120day.ncm

NAFCOM 96 COST MODEL
WBS COST
1999 M$

Prepared By:
Revision:

	WBS Element	Prod FBS Factor	D&D Cmplx.	D&D Inher.	Unit Cmplx.	STH Qty.	STH %	Weight	Learning %	Prod Start Unit	Prod Rate/Year
1	GRAND TOTAL	1.000						329.4010000000000104			
2	SYSTEM 1: Free Flying Servicer	1.000						329.4010000000000104			
3	HARDWARE TOTAL	1.000						329.4010000000000104			
4	R.S. Propulsion	1.000						22			
5	Propulsion Unit (Chemical)	1.000	1.00	1.00	1.00	1	130	22	90	1	1
6	Payload	1.000						61.40100000000000336			
7	microsatellite	1.000						61.40000000000000568			
8	structure & mechanical	1.000	1.00	1.00	1.00	1	130	24.6	90	1	1
9	Avionics	1.000	1.00	1.00	1.00	1	130	28	90	1	1
10	Power system (batteries)	1.000	1.00	1.00	1.00	1	130	8.8	90	1	1
11	Arms & End effectors	1.000	1.00	1.00	1.00	1	130	.001	90	1	1
12	Spacecraft Bus	1.000						246			
13	Thermal control	1.000	1.00	1.00	1.00	1	130	10	90	1	1
14	Structures & Mechanisms	1.000	1.00	1.00	1.00	1	130	75	90	1	1
15	ADAC's & RCS	1.000	1.00	1.00	1.00	1	130	86	90	1	1
16	Data Management System	1.000	1.00	1.00	1.00	1	130	4	90	1	1
17	Communication System	1.000	1.00	1.00	1.00	1	130	10	90	1	1
18	Electrical Power System	1.000	1.00	1.00	1.00	1	130	61	90	1	1
19	SYSTEM INTEGRATION SUBTOTAL	1.000									
20	IACO		1		1						
21	STO		1								
22	GSE										
23	Tooling		1								
24	M/E GSE		1								
25	SE&I		1		1						
26	PM		1		1						
27	LOOS		1								
28	Contingency										
29	Program Support										
30	Fee										
31											
32											
33	FUNCTIONAL RATES (FY96$)										
34	Engineering Labor Hourly Rate: $34.00										
35	Manufacturing Labor Hourly Rate: $28.00										

Appendix J: NAFCOM Cost Sheet – Low Capability, 2 Year Design Life RS

Project: New Estimate
File Name: C:\NAFCOM96\Low_R.S._2year.ncm

NAFCOM 96 COST MODEL
WBS COST
1999 M$

Prepared By:
Revision:

	WBS Element	DDT&E	D&D	STH	Flight Unit	Production	Total	DDT&E FBS Factor	D&D FBS Factor	STH FBS Factor	FU FBS Factor
1	GRAND TOTAL	42.5			9.5	48.4	90.9	1.000			1.000
2	SYSTEM 1: Free Flying Servicer	42.5			9.5	48.4	90.9	1.000			1.000
3	HARDWARE TOTAL	26.2	17.7	8.5	6.6	33.3	59.5		1.000	1.000	1.000
4	R.S. Propulsion	0.4	0.3	0.2	0.1	0.7	1.1		1.000	1.000	1.000
5	Propulsion Unit (Chemical)	0.4	0.3	0.2	0.1	0.7	1.1		1.000	1.000	1.000
6	Payload	6.1	4.1	2.0	1.5	7.9	14.0		1.000	1.000	1.000
7	microsatellite	6.1	4.1	2.0	1.5	7.9	14.0		1.000	1.000	1.000
8	structure & mechanical	1.1	1.0	0.2	0.1	0.8	1.9		1.000	1.000	1.000
9	Avionics	4.1	2.6	1.5	1.2	6.0	10.1		1.000	1.000	1.000
10	Power system (batteries)	0.9	0.6	0.3	0.2	1.1	2.0		1.000	1.000	1.000
11	Arms & End effectors	0.0	0.0	0.0	0.0	0.0	0.0		1.000	1.000	1.000
12	Spacecraft Bus	19.7	13.3	6.3	4.9	24.7	44.4		1.000	1.000	1.000
13	Thermal control	1.1	0.9	0.2	0.2	0.8	1.8		1.000	1.000	1.000
14	Structures & Mechanisms	2.1	1.6	0.5	0.4	2.0	4.1		1.000	1.000	1.000
15	ADAC's & RCS	10.4	6.9	3.4	2.6	13.3	23.7		1.000	1.000	1.000
16	Data Management System	1.7	1.1	0.6	0.4	2.2	3.9		1.000	1.000	1.000
17	Communication System	0.6	0.3	0.3	0.2	1.2	1.9		1.000	1.000	1.000
18	Electrical Power System	3.8	2.5	1.3	1.0	5.2	9.0		1.000	1.000	1.000
19	SYSTEM INTEGRATION SUBTOTAL	16.2			3.0	15.1	31.3	1.000			1.000
20	IACO	1.9			0.8	4.2	6.1				
21	STO	0.9			N/A	N/A	0.9				
22	GSE										
23	Tooling	0.4			N/A	N/A	0.4				
24	M/E GSE	3.3			N/A	N/A	3.3				
25	SE&I	5.6			1.3	6.7	12.4				
26	PM	2.8			0.8	4.2	7.1				
27	LOOS	1.3			N/A	N/A	1.3				
28	Contingency	0.0			0.0	0.0	0.0				
29	Program Support	0.0			0.0	0.0	0.0				
30	Fee	0.0			0.0	0.0	0.0				
31											
32											
33	FUNCTIONAL RATES (FY96$)										
34	Engineering Labor Hourly Rate: $34.00										
35	Manufacturing Labor Hourly Rate: $28.00										

Appendix K: NAFCOM Cost Sheet – Low Capability, 15 Year Design Life RS

Project: New Estimate
File Name: C:\NAFCOM96\Low_R.S._15 year.ncm

NAFCOM 96 COST MODEL
WBS COST
1999 M$

Prepared By:
Revision:

	WBS Element	DDT&E	D&D	STH	Flight Unit	Production	Total	DDT&E FBS Factor	D&D FBS Factor	STH FBS Factor	FU FBS Factor
1	GRAND TOTAL	52.6			11.9	60.1	112.7	1.000			1.000
2	SYSTEM 1: Free Flying Servicer	52.6			11.9	60.1	112.7	1.000			1.000
3	HARDWARE TOTAL	32.8	22.1	10.7	8.2	41.7	74.5		1.000	1.000	1.000
4	R.S. Propulsion	1.9	1.1	0.8	0.6	3.1	5.0		1.000	1.000	1.000
5	Propulsion Unit (Chemical)	1.9	1.1	0.8	0.6	3.1	5.0		1.000	1.000	1.000
6	Payload	8.5	5.9	2.6	2.0	10.1	18.6		1.000	1.000	1.000
7	microsatellite	7.1	4.8	2.3	1.8	9.2	16.3		1.000	1.000	1.000
8	structure & mechanical	1.3	1.1	0.2	0.2	0.9	2.2		1.000	1.000	1.000
9	Avionics	4.8	3.0	1.8	1.4	7.0	11.8		1.000	1.000	1.000
10	Power system (batteries)	1.0	0.7	0.3	0.3	1.3	2.3		1.000	1.000	1.000
11	Arms & End effectors	1.4	1.1	0.2	0.2	0.9	2.3		1.000	1.000	1.000
12	Spacecraft Bus	22.4	15.0	7.3	5.6	28.5	50.9		1.000	1.000	1.000
13	Thermal control	1.2	0.9	0.2	0.2	0.9	2.1		1.000	1.000	1.000
14	Structures & Mechanisms	2.1	1.6	0.5	0.4	2.1	4.2		1.000	1.000	1.000
15	ADAC's & RCS	11.9	8.0	4.0	3.1	15.5	27.5		1.000	1.000	1.000
16	Data Management System	2.0	1.3	0.7	0.5	2.5	4.5		1.000	1.000	1.000
17	Communication System	0.7	0.4	0.4	0.3	1.4	2.2		1.000	1.000	1.000
18	Electrical Power System	4.4	2.9	1.6	1.2	6.1	10.5		1.000	1.000	1.000
19	SYSTEM INTEGRATION SUBTOTAL	19.9			3.6	18.4	38.2	1.000			1.000
20	IACO	2.3			1.0	4.9	7.2				
21	STO	1.0			N/A	N/A	1.0				
22	GSE										
23	Tooling	0.5			N/A	N/A	0.5				
24	M/E GSE	4.2			N/A	N/A	4.2				
25	SE&I	7.0			1.6	8.4	15.4				
26	PM	3.4			1.0	5.1	8.5				
27	LOOS	1.6			N/A	N/A	1.6				
28	Contingency	0.0			0.0	0.0	0.0				
29	Program Support	0.0			0.0	0.0	0.0				
30	Fee	0.0			0.0	0.0	0.0				
31											
32											
33	FUNCTIONAL RATES (FY96$)										
34	Engineering Labor Hourly Rate: $34.00										
35	Manufacturing Labor Hourly Rate: $28.00										

 NAFCOM COST MODEL CONTROL NUMBER: NCM0139 02/18/99 09:42:29

Appendix L: NAFCOM Cost Sheet – Medium Capability, 120 Day Design Life RS

Project: New Estimate
File Name: C:\NAFCOM96\Med_R.S_120day.ncm

NAFCOM 96 COST MODEL
WBS COST
1999 M$

Prepared By:
Revision:

	WBS Element	DDT&E	D&D	STH	Flight Unit	Production	Total	DDT&E FBS Factor	D&D FBS Factor	STH FBS Factor	FU FBS Factor
1	GRAND TOTAL	55.6			15.7	79.7	135.3	1.000			1.000
2	SYSTEM 1: Operational Ranger Servicer	55.6			15.7	79.7	135.3	1.000			1.000
3	HARDWARE TOTAL	34.4	20.0	14.4	11.0	56.0	90.4		1.000	1.000	1.000
4	R.S. Propulsion	2.2	1.3	0.9	0.7	3.6	5.8		1.000	1.000	1.000
5	Propulsion Unit (Chemical)	2.2	1.3	0.9	0.7	3.6	5.8		1.000	1.000	1.000
6	Payload	9.8	3.7	6.1	4.7	23.7	33.5		1.000	1.000	1.000
7	Arms & End effectors	9.8	3.7	6.1	4.7	23.7	33.5		1.000	1.000	1.000
8	Spacecraft Bus	22.4	15.1	7.4	5.7	28.7	51.1		1.000	1.000	1.000
9	Thermal control	1.0	0.8	0.2	0.1	0.7	1.7		1.000	1.000	1.000
10	Structures & Mechanisms	2.1	1.6	0.5	0.4	2.0	4.0		1.000	1.000	1.000
11	ADAC's & RCS	13.7	9.1	4.6	3.6	18.0	31.7		1.000	1.000	1.000
12	Data Management System	1.4	0.9	0.5	0.4	1.8	3.2		1.000	1.000	1.000
13	Communication System	0.6	0.3	0.3	0.2	1.2	1.9		1.000	1.000	1.000
14	Electrical Power System	3.6	2.4	1.2	1.0	4.9	8.5		1.000	1.000	1.000
15	SYSTEM INTEGRATION SUBTOTAL	21.2			4.7	23.7	44.9	1.000			1.000
16	IACO	2.8			1.2	6.1	8.9				
17	STO	1.0			N/A	N/A	1.0				
18	GSE										
19	Tooling	0.5			N/A	N/A	0.5				
20	M/E GSE	4.4			N/A	N/A	4.4				
21	SE&I	7.4			2.2	11.0	18.4				
22	PM	3.5			1.3	6.5	10.0				
23	LOOS	1.7			N/A	N/A	1.7				
24	Contingency	0.0			0.0	0.0	0.0				
25	Program Support	0.0			0.0	0.0	0.0				
26	Fee	0.0			0.0	0.0	0.0				
27											
28											
29	FUNCTIONAL RATES (FY96$)										
30	Engineering Labor Hourly Rate: $34.00										
31	Manufacturing Labor Hourly Rate: $28.00										
32	Other Labor Hourly Rate: $31.00										
33	Overhead Rate: 135.00 %										
34	G & A Rate: 10.00 %										

Appendix M: NAFCOM Cost Sheet – Medium Capability, 2 Year Design Life RS

Project: New Estimate
File Name: C:\NAFCOM96\Med_R.S_2year.ncm

NAFCOM 96 COST MODEL
WBS COST
1999 M$

Prepared By:
Revision:

	WBS Element	DDT&E	D&D	STH	Flight Unit	Production	Total	DDT&E FBS Factor	D&D FBS Factor	STH FBS Factor	FU FBS Factor
1	GRAND TOTAL	55.8			15.8	79.9	135.7	1.000			1.000
2	SYSTEM 1: Operational Ranger Servicer	55.8			15.8	79.9	135.7	1.000			1.000
3	HARDWARE TOTAL	34.5	20.1	14.4	11.1	56.1	90.6		1.000	1.000	1.000
4	R.S. Propulsion	0.6	0.3	0.2	0.2	0.8	1.4		1.000	1.000	1.000
5	Propulsion Unit (Chemical)	0.6	0.3	0.2	0.2	0.8	1.4		1.000	1.000	1.000
6	Payload	10.1	3.8	6.3	4.9	24.6	34.7		1.000	1.000	1.000
7	Arms & End effectors	10.1	3.8	6.3	4.9	24.6	34.7		1.000	1.000	1.000
8	Spacecraft Bus	23.8	16.0	7.9	6.0	30.7	54.5		1.000	1.000	1.000
9	Thermal control	1.1	0.9	0.2	0.2	0.8	1.8		1.000	1.000	1.000
10	Structures & Mechanisms	2.1	1.6	0.5	0.4	2.0	4.1		1.000	1.000	1.000
11	ADAC's & RCS	14.2	9.4	4.8	3.7	18.7	32.9		1.000	1.000	1.000
12	Data Management System	1.7	1.1	0.6	0.4	2.2	3.9		1.000	1.000	1.000
13	Communication System	0.6	0.3	0.3	0.2	1.2	1.9		1.000	1.000	1.000
14	Electrical Power System	4.2	2.7	1.5	1.1	5.7	9.9		1.000	1.000	1.000
15	SYSTEM INTEGRATION SUBTOTAL	21.3			4.7	23.8	45.0	1.000			1.000
16	IACO	2.8			1.2	6.2	8.9				
17	STO	1.0			N/A	N/A	1.0				
18	GSE										
19	Tooling	0.5			N/A	N/A	0.5				
20	M/E GSE	4.4			N/A	N/A	4.4				
21	SE&I	7.4			2.2	11.1	18.5				
22	PM	3.5			1.3	6.6	10.1				
23	LOOS	1.7			N/A	N/A	1.7				
24	Contingency	0.0			0.0	0.0	0.0				
25	Program Support	0.0			0.0	0.0	0.0				
26	Fee	0.0			0.0	0.0	0.0				
27											
28											
29	FUNCTIONAL RATES (FY96$)										
30	Engineering Labor Hourly Rate: $34.00										
31	Manufacturing Labor Hourly Rate: $28.00										
32	Other Labor Hourly Rate: $31.00										
33	Overhead Rate: 135.00 %										
34	G & A Rate: 10.00 %										

Appendix N: NAFCOM Cost Sheet – Medium Capability, 15 Year Design Life RS

Project: New Estimate
File Name: C:\NAFCOM96\Med_R.S_15year.ncm

NAFCOM 96 COST MODEL
WBS COST
1999 M$

Prepared By:
Revision:

	WBS Element	DDT&E	D&D	STH	Flight Unit	Production	Total	DDT&E FBS Factor	D&D FBS Factor	STH FBS Factor	FU FBS Factor
1	GRAND TOTAL	64.6			18.1	91.8	156.4	1.000			1.000
2	SYSTEM 1: Operational Ranger Servicer	64.6			18.1	91.8	156.4	1.000			1.000
3	HARDWARE TOTAL	40.2	23.6	16.6	12.8	64.8	105.0		1.000	1.000	1.000
4	R.S. Propulsion	2.4	1.4	1.0	0.8	4.0	6.4		1.000	1.000	1.000
5	Propulsion Unit (Chemical)	2.4	1.4	1.0	0.8	4.0	6.4		1.000	1.000	1.000
6	Payload	10.1	3.8	6.3	4.9	24.6	34.7		1.000	1.000	1.000
7	Arms & End effectors	10.1	3.8	6.3	4.9	24.6	34.7		1.000	1.000	1.000
8	Spacecraft Bus	27.7	18.4	9.3	7.1	36.2	63.9		1.000	1.000	1.000
9	Thermal control	1.2	0.9	0.2	0.2	0.9	2.1		1.000	1.000	1.000
10	Structures & Mechanisms	2.1	1.6	0.5	0.4	2.1	4.2		1.000	1.000	1.000
11	ADAC's & RCS	16.4	10.8	5.6	4.3	21.9	38.2		1.000	1.000	1.000
12	Data Management System	2.0	1.3	0.7	0.5	2.5	4.5		1.000	1.000	1.000
13	Communication System	0.7	0.4	0.4	0.3	1.4	2.2		1.000	1.000	1.000
14	Electrical Power System	5.3	3.4	1.9	1.5	7.4	12.7		1.000	1.000	1.000
15	SYSTEM INTEGRATION SUBTOTAL	24.4			5.3	27.0	51.4	1.000			1.000
16	IACO	3.1			1.4	6.9	9.9				
17	STO	1.1			N/A	N/A	1.1				
18	GSE										
19	Tooling	0.6			N/A	N/A	0.6				
20	M/E GSE	5.1			N/A	N/A	5.1				
21	SE&I	8.6			2.5	12.7	21.3				
22	PM	3.9			1.5	7.4	11.3				
23	LOOS	2.0			N/A	N/A	2.0				
24	Contingency	0.0			0.0	0.0	0.0				
25	Program Support	0.0			0.0	0.0	0.0				
26	Fee	0.0			0.0	0.0	0.0				
27											
28											
29	FUNCTIONAL RATES (FY96$)										
30	Engineering Labor Hourly Rate: $34.00										
31	Manufacturing Labor Hourly Rate: $28.00										
32	Other Labor Hourly Rate: $31.00										
33	Overhead Rate: 135.00 %										
34	G & A Rate: 10.00 %										

Appendix O: NAFCOM Cost Sheet – High Capability, 120 Day Design Life RS

Project: New Estimate
File Name: C:\NAFCOM96\Operational_Ranger.ncm

NAFCOM 96 COST MODEL
WBS COST
1998 M$

Prepared By:
Revision:

	WBS Element	DDT&E	D&D	STH	Flight Unit	Production	Total	DDT&E FBS Factor	D&D FBS Factor	STH FBS Factor	FU FBS Factor
1	GRAND TOTAL	77.6			20.5	104.1	181.7	1.000			1.000
2	SYSTEM 1: Operational Ranger Servicer	77.6			20.5	104.1	181.7	1.000			1.000
3	HARDWARE TOTAL	48.5	29.7	18.9	14.5	73.7	122.2		1.000	1.000	1.000
4	R.S. Propulsion	3.4	2.0	1.4	1.0	5.3	8.7		1.000	1.000	1.000
5	Propulsion Unit (Chemical)	3.4	2.0	1.4	1.0	5.3	8.7		1.000	1.000	1.000
6	Payload	9.4	3.6	5.8	4.5	22.6	32.0		1.000	1.000	1.000
7	Arms & End effectors	9.4	3.6	5.8	4.5	22.6	32.0		1.000	1.000	1.000
8	Spacecraft Bus	35.8	24.1	11.7	9.0	45.8	81.6		1.000	1.000	1.000
9	Thermal control	1.8	1.4	0.4	0.3	1.4	3.2		1.000	1.000	1.000
10	Structures & Mechanisms	3.3	2.5	0.8	0.6	3.2	6.5		1.000	1.000	1.000
11	ADAC's & RCS	19.7	13.1	6.5	5.0	25.5	45.1		1.000	1.000	1.000
12	Data Management System	3.2	2.2	1.1	0.8	4.1	7.3		1.000	1.000	1.000
13	Communication System	1.1	0.5	0.6	0.4	2.3	3.4		1.000	1.000	1.000
14	Electrical Power System	6.7	4.3	2.4	1.8	9.3	16.0		1.000	1.000	1.000
15	SYSTEM INTEGRATION SUBTOTAL	29.1			6.0	30.5	59.5	1.000			1.000
16	IACO	3.5			1.5	7.7	11.2				
17	STO	1.3			N/A	N/A	1.3				
18	GSE										
19	Tooling	0.7			N/A	N/A	0.7				
20	M/E GSE	6.2			N/A	N/A	6.2				
21	SE&I	10.4			2.8	14.4	24.8				
22	PM	4.6			1.7	8.4	13.0				
23	LOOS	2.5			N/A	N/A	2.5				
24	Contingency	0.0			0.0	0.0	0.0				
25	Program Support	0.0			0.0	0.0	0.0				
26	Fee	0.0			0.0	0.0	0.0				
27											
28											
29	FUNCTIONAL RATES (FY96$)										
30	Engineering Labor Hourly Rate: $34.00										
31	Manufacturing Labor Hourly Rate: $28.00										
32	Other Labor Hourly Rate: $31.00										
33	Overhead Rate: 135.00 %										
34	G & A Rate: 10.00 %										

Appendix P: NAFCOM Cost Sheet – High Capability, 2 Year Design Life RS

Project: New Estimate
File Name: C:\NAFCOM96\Operational_Ranger_2year.ncm

NAFCOM 96 COST MODEL
WBS COST
1998 M$

Prepared By:
Revision:

	WBS Element	DDT&E	D&D	STH	Flight Unit	Production	Total	DDT&E FBS Factor	D&D FBS Factor	STH FBS Factor	FU FBS Factor
1	GRAND TOTAL	92.2			26.4	134.0	226.2	1.000			1.000
2	SYSTEM 1: Operational Ranger Servicer	92.2			26.4	134.0	226.2	1.000			1.000
3	HARDWARE TOTAL	57.9	33.3	24.5	18.9	95.7	153.5		1.000	1.000	1.000
4	R.S. Propulsion	0.9	0.5	0.3	0.2	1.3	2.1		1.000	1.000	1.000
5	Propulsion Unit (Chemical)	0.9	0.5	0.3	0.2	1.3	2.1		1.000	1.000	1.000
6	Payload	16.7	5.9	10.8	8.3	42.3	59.0		1.000	1.000	1.000
7	Arms & End effectors	16.7	5.9	10.8	8.3	42.3	59.0		1.000	1.000	1.000
8	Spacecraft Bus	40.3	26.9	13.4	10.3	52.1	92.4		1.000	1.000	1.000
9	Thermal control	1.9	1.5	0.4	0.3	1.6	3.4		1.000	1.000	1.000
10	Structures & Mechanisms	3.9	2.9	1.0	0.8	3.9	7.8		1.000	1.000	1.000
11	ADAC's & RCS	21.4	14.2	7.2	5.5	28.0	49.4		1.000	1.000	1.000
12	Data Management System	3.5	2.4	1.2	0.9	4.5	8.1		1.000	1.000	1.000
13	Communication System	1.2	0.6	0.6	0.5	2.5	3.7		1.000	1.000	1.000
14	Electrical Power System	8.3	5.3	3.0	2.3	11.7	20.0		1.000	1.000	1.000
15	SYSTEM INTEGRATION SUBTOTAL	34.4			7.6	38.3	72.7	1.000			1.000
16	IACO	4.1			1.8	9.4	13.5				
17	STO	1.4			N/A	N/A	1.4				
18	GSE										
19	Tooling	0.8			N/A	N/A	0.8				
20	M/E GSE	7.4			N/A	N/A	7.4				
21	SE&I	12.4			3.6	18.4	30.8				
22	PM	5.3			2.1	10.5	15.7				
23	LOOS	3.0			N/A	N/A	3.0				
24	Contingency	0.0			0.0	0.0	0.0				
25	Program Support	0.0			0.0	0.0	0.0				
26	Fee	0.0			0.0	0.0	0.0				
27											
28											
29	FUNCTIONAL RATES (FY96$)										
30	Engineering Labor Hourly Rate: $34.00										
31	Manufacturing Labor Hourly Rate: $28.00										
32	Other Labor Hourly Rate: $31.00										
33	Overhead Rate: 135.00 %										
34	G & A Rate: 10.00 %										

Appendix Q: NAFCOM Cost Sheet – High Capability, 15 Year Design Life RS

Project: New Estimate
File Name: C:\NAFCOM96\Operational_Ranger_15year.ncm

NAFCOM 96 COST MODEL
WBS COST
1998 M$

Prepared By:
Revision:

	WBS Element	DDT&E	D&D	STH	Flight Unit	Production	Total	DDT&E FBS Factor	D&D FBS Factor	STH FBS Factor	FU FBS Factor
1	GRAND TOTAL	111.4			32.1	162.7	274.1	1.000			1.000
2	SYSTEM 1: Operational Ranger Servicer	111.4			32.1	162.7	274.1	1.000			1.000
3	HARDWARE TOTAL	70.3	40.3	30.0	23.1	117.0	187.3		1.000	1.000	1.000
4	R.S. Propulsion	4.2	2.5	1.7	1.3	6.6	10.7		1.000	1.000	1.000
5	Propulsion Unit (Chemical)	4.2	2.5	1.7	1.3	6.6	10.7		1.000	1.000	1.000
6	Payload	18.8	6.5	12.3	9.5	48.0	66.8		1.000	1.000	1.000
7	Arms & End effectors	18.8	6.5	12.3	9.5	48.0	66.8		1.000	1.000	1.000
8	Spacecraft Bus	47.3	31.3	16.0	12.3	62.4	109.7		1.000	1.000	1.000
9	Thermal control	2.1	1.7	0.5	0.4	1.8	4.0		1.000	1.000	1.000
10	Structures & Mechanisms	4.0	3.0	1.0	0.8	4.0	8.0		1.000	1.000	1.000
11	ADAC's & RCS	24.7	18.3	8.4	6.4	32.6	57.3		1.000	1.000	1.000
12	Data Management System	4.2	2.8	1.4	1.0	5.3	9.5		1.000	1.000	1.000
13	Communication System	1.4	0.7	0.7	0.6	2.9	4.3		1.000	1.000	1.000
14	Electrical Power System	10.9	6.8	4.0	3.1	15.8	26.6		1.000	1.000	1.000
15	SYSTEM INTEGRATION SUBTOTAL	41.1			9.0	45.7	86.8	1.000			1.000
16	IACO	4.8			2.2	10.9	15.7				
17	STO	1.6			N/A	N/A	1.6				
18	GSE										
19	Tooling	1.0			N/A	N/A	1.0				
20	M/E GSE	8.9			N/A	N/A	8.9				
21	SE&I	15.0			4.4	22.3	37.4				
22	PM	6.1			2.4	12.4	18.5				
23	LOOS	3.7			N/A	N/A	3.7				
24	Contingency	0.0			0.0	0.0	0.0				
25	Program Support	0.0			0.0	0.0	0.0				
26	Fee	0.0			0.0	0.0	0.0				
27											
28											
29	FUNCTIONAL RATES (FY96$)										
30	Engineering Labor Hourly Rate: $34.00										
31	Manufacturing Labor Hourly Rate: $28.00										
32	Other Labor Hourly Rate: $31.00										
33	Overhead Rate: 135.00 %										
34	G & A Rate: 10.00 %										

Appendix R: NAFCOM Cost Sheet for Ion Propulsion

Project: New Estimate
File Name: C:\NAFCOM96\ion propulsion system.ncm

NAFCOM 96 COST MODEL
WBS COST
1999 M$

Prepared By:
Revision:

	WBS Element	Prod FBS Factor	D&D Cmplx.	D&D Inher.	Unit Cmplx.	STH Qty.	STH %	Weight	Learning %	Prod Start Unit	Prod Rate/Year	Rate %
1	GRAND TOTAL	1.000						784				
2	SYSTEM 1:	1.000						784				
3	HARDWARE TOTAL	1.000						784				
4	solar arrays for electric propulsion alternatives	1.000	1.00	1.00	1.00	1	130	183	100	1	1	100
5	ion propulsion system	1.000	1.00	1.00	1.00	1	130	601	100	1	1	100
6	SYSTEM INTEGRATION SUBTOTAL	1.000										
7	IACO		0		0							
8	STO		0									
9	GSE											
10	Tooling		0									
11	M/E GSE		0									
12	SE&I		0		0							
13	PM		0		0							
14	LOOS		0									
15	Contingency											
16	Program Support											
17	Fee											
18												
19												
20	FUNCTIONAL RATES (FY96$)											
21	Engineering Labor Hourly Rate: $34.00											
22	Manufacturing Labor Hourly Rate: $28.00											
23	Other Labor Hourly Rate: $31.00											
24	Overhead Rate: 135.00 %											
25	G & A Rate: 10.00 %											

Project: New Estimate
File Name: C:\NAFCOM96\ion propulsion system.ncm

NAFCOM 96 COST MODEL
WBS COST
1999 M$

Prepared By:
Revision:

	WBS Element	LRIP Step Down %	LRIP Quantity	QNHA	Make %
1	GRAND TOTAL				
2	SYSTEM 1:				
3	HARDWARE TOTAL				
4	solar arrays for electric propulsion alternatives	5	0	1	75
5	Ion propulsion system	5	0	1	75
6	SYSTEM INTEGRATION SUBTOTAL				100
7	IACO				
8	STO				
9	GSE				
10	Tooling				
11	M/E GSE				
12	SE&I				
13	PM				
14	LOOS				
15	Contingency				
16	Program Support				
17	Fee				
18					
19					
20	FUNCTIONAL RATES (FY96$)				
21	Engineering Labor Hourly Rate: $34.00				
22	Manufacturing Labor Hourly Rate: $28.00				
23	Other Labor Hourly Rate: $31.00				
24	Overhead Rate: 135.00 %				
25	G & A Rate: 10.00 %				

Appendix S: Detailed Description of AweSim Model

First Loop – Detailed Description

We use AweSim terminology to describe the simulation. See Simulation with Visual SLAM and AweSim for explanations of the simulation components and terminology. We also intend for the reader to follow along in the printout of the simulation while reading this. Thus, when we use the words "above" or "below," we are referring to relative positions in the simulation structure.

At the top of the simulation are six RESOURCES. Each RESOURCE represents the capacity for a servicer to exist in each plane. Loop two gives these RESOURCES a unit of availability in accordance with the number of servicers for that scenario, the need and the review cycle. The CREATE node at the beginning of the first loop generates 33 satellite entities. The interarrival times are drawn from an array statement in the control file. The "PlaneSet" ASSIGN node places each satellite at random into one of the six planes. This is to simulate an unknown pattern of Block IIR failures and corresponding replacements with IIF satellites. The conditions on the attached ACTIVITIES ensure that no plane gets too many satellites. The following "SatLbl" ASSIGN node gives each satellite its SVN (Satellite Vehicle Number), increments the number of satellites in that satellite's plane, and the final assignment sets a value of 1 to indicate a live satellite.

The line that starts with the "Cont1" GOON node generates the clones that simulate different portions of the satellite. The first clone tracks the number of active satellites in the system. These entities wait in the "SatTrack" QUEUE. A FINDAR node at the end of loop one pulls and reroutes these entities as soon as any clones of a satellite leave the system at the end of this loop. The next clone tracks unrepairable failures. The

delay on this activity is a draw from the associated random distribution. When the clone completes the activity, the "Fail1" ASSIGN node sets the global variable tracking the satellite's status to zero indicating a dead satellite. The "ErrType" and "ErrLabel" are used in condition statements and collect nodes later in the simulation. Before sending the entity to the "EndFail" GOON node, the branching helps to collect data on extending the lives of the satellites. There were 33 elements in row 2 of the ARRAY to correspond to each SVN. The control file initializes the array values to 0. Those zeroes take on the TNOW value the first time a live satellite experiences a failure in a repairable system. When the satellite experiences an unrepairable failure, XX[52] adds up the satellite longevity extensions. At the end of the simulation, a collect node calculates the average life extension. The next clone models array failures. The distribution on array failure comes from Dr. Womack's paper entitled Revised Block II/IIA Lifetime Predictions and the Impact on Block IIR/IIF Replenishment Planning. LL[3] is the amount of advance time that the review loop uses to guard against unrepaired failures. We assumed that GPS personnel would monitor degrading components and be able to predict failure in advance by the desired amount of time. The last clone models clock failures through the final level of redundancy. These failures are also predicted in advance for the purposes of the review loop. This failure data also came from Dr. Womack's paper. He had the individual clock failure distributions, and we used Excel to stochastically determine combined failure data. From that we used the Best Fit software to find the new distribution. We talked to Dr. Womack about those results, and he concurred. Each individual failure mode clone passes through an ASSIGN node and then a write node to track failures. I used the write nodes during the troubleshooting process. The "Refresh1"

ASSIGN node that each repairable failure clone passes through resets a variable used later in the simulation.

The "FixIt" GOON node begins the process of sorting the unrepairable clones to trigger servicing actions. The condition on the first activity catches clones from satellites that have already suffered an unrepairable failure or bypasses the servicing possibilities for the baseline scenario when the number of available servicers is zero. The "Cont2" GOON node sends the clones through the appropriate activities to complete the life of that subsystem and then sends the clone to the conclusion of the first loop. The next activity from the "FixIt" node takes the first clone of a live satellite that comes through and records the time. The "Cont4" node sorts the clones, and ATRIB[2] stores the future time of failure for that subsystem. Then, the clones enter the "Anteroom" AWAIT node to wait for loop two to open the corresponding "Review" GATE. When each review cycle occurs, the gate opens and increments a global variable that tracks the number of serviceable failures in the plane. The purpose of the "AnteRoom" AWAIT node and the GATE is to prevent failures that occur between review cycles from being seen by the servicer. This is because the failures should presumably not be seen by the servicer until the review board identifies them during the next cycle. After incrementing the failure count for the necessary planes, the clones enter the "Wait1" AWAIT node for loop two to give the appropriate RESOURCE a unit of availability. When the clones leave the AWAIT node, they go to COLLECT node "Chk1" or "Chk2" according to whether or not the current time is after the failure time of the subsystem or before the failure time of the subsystem respectively. On the way to "Chk1" the clone passes through an ASSIGN

node. It adds to the XX[60] variable the length of the gap, and increments XX[61], which counts the number of late servicings.

The "Chk1" COLLECT node tracks the gap in time after subsystem failure that service occurs. The "Chk2" COLLECT node tracks the spare time prior to subsystem failure that service occurs. At the "Cont3" GOON node, clones for satellites that have already failed are routed through a WRITE node and then free the RESOURCE those clones acquired. If an unrepairable failure for the corresponding satellite has not occurred when the clone reaches the "Cont3" node, the satellite subsystem receives service. The "ServSat" ASSIGN node increments LL[5], which counts the number of servicing missions. When the simulation ends, XX[55] will hold the time of the last service. The third assignment individually tracks the number of services for each repairable subsystem. The delay on the following activity is the length of time to perform a servicing mission. The RESOURCE is then freed to perform other servicing missions in that plane if necessary. After the "ResFree1" FREE node, the clone passes through the "Fixed" WRITE node, and then all clones decrement the number of failures in that plane. The clones continue to the "Sort1" GOON node.

The primary purpose of this next portion of loop one is to decrement the availability of the servicer resource as appropriate and to refresh the longevity of the subsystems that receive service. The first activity takes clones when there are no remaining failures needing repair in that clone's plane. The "Unavail1" ALTER node then reduces the availability of that plane's RESOURCE by one to zero. LL[20] tracks the number of active servicers, so the "RSdone1" node decrements that value. At the "Split1" GOON node, clones of dead satellites take the first activity and the rest take the

second activity. At the "Sort1" node, if a clone was not the last failure in the plane but is part of a dead satellite, it takes the next activity. Clones that come to the "Split2" GOON node take the first emanating activity if the current time is less than the time of failure for that subsystem. This is to prevent underestimation of subsystem failure times for clones of satellites that suffer unrepairable failures after the satellite has already be slated for a subsystem repair. Such failures can occur during orbit to orbit transfer times in scenarios with one servicer. Overestimation is still possible, but this did not impact the statistics in which we were primarily interested. All clones that represent subsystems receiving repair on an otherwise healthy satellite continue to the "Sort2" GOON node. They continue along the appropriate activity and the longevity for that subsystem is refreshed. The simulation then reroutes the clone to the corresponding write node above.

The "EndFail" GOON node begins the final portion of loop one. All failure mode clones arrive here. In scenarios with servicers, repairable failure clones arrive here only after the unrepairable failure clones terminate. In scenarios without servicers, all failure mode clones arrive here as soon as they have completed their failure mode delay in the "Cont1" portion of the simulation above. The first emanating activity takes only clones that represent the un-repairable portion of the satellite. The "FailSat1" COLLECT node gathers data on the length of time the clone was in the system. The following ASSIGN node sets global variables for the subsequent FINDAR nodes. From the necessary AWAIT nodes above, they extracted the clones corresponding to the unrepairable failure clone that triggered the FINDAR nodes. The "PullSat1" FINDAR node extracts clones from the file corresponding to that satellite's plane. Clones in this position represent repairable failure modes that the loop two review cycle has already tagged for repair but

are still awaiting service. The FINDAR node pulls the clone from the file in which it is waiting and routes it to the "Detour1" ASSIGN node. The ASSIGN node decrements the failure count in that plane, and it makes sure that the count doesn't go below zero. This precaution was necessary due to overlap of events that happen close together in the same plane. The "PullSat2" FINDAR node pulls corresponding clones from the "AnteRoom" AWAIT node above. In both FINDAR nodes, they do not reroute clones if the servicer has just arrived (RESOURCE has just been activated) to their plane. This prevents the situation where a servicer is en route to a satellite that disappears before the servicer arrives. When the servicer arrives, it checks to make sure the subsystem still warrants service. That occurs at the "Cont3" GOON node above. The clone then goes to the "Tracker" WRITE node. The other failure mode clones that enter the "EndFail" GOON node take the second activity. Each failure type has an associated COLLECT node to gather longevity data for each subsystem. The "Tracker" node writes all the failure data to a file called "output.txt." The "Num1" ASSIGN node and "PullSat3" FINDAR node help track the number of operational satellites in the system. The first time that any clone for a satellite passes through these nodes, the FINDAR pulls from the "SatTrack" QUEUE the clone that went along the first activity node from the "Cont1" GOON node. The FINDAR routes that clone to the "DETOUR3" ASSIGN node in loop three for statistics collection purposes. The failure mode clone then leaves the system through the "DeadSat" TERMINATE node. This completes loop one.

Second Loop – Detailed Description

The second loop simulates a review board that examines the constellation of satellites for repairable failures, considers the location of the servicer or servicers and

sends the servicers on servicing missions as appropriate. The "BootServ" CREATE node is the start point of loop two. It creates entities where each entity represents a convening of a review board. "BootServ" creates entities starting at time 0. The time between creations is the time between review cycles as appropriate for the current scenario. The "ChkNeed" QUEUE is a leftover from previous iterations of the simulation design but does not interfere with the simulation. The review cycle entity's first action is to trigger the "Release" OPEN node. This opens the GATE holding shut the "AnteRoom" AWAIT node in loop one. The delay on the following activity ensures that closing the GATE does not occur instantaneously. Then, if any plane has failures in need of repair, the review entity takes the first activity to the "Chk3" GOON node. If no plane is in need of attention, the entity goes to the "Done1" GOON node, closes the gate and terminates. From the "Chk3" node, the entity takes the first activity if the number of currently active servicers (LL[20]) equals the number of available servicers (LL[2]). Otherwise, the entity continues to the "Sort3" node. The entity encounters six pairs of conditioned activities and a final unconditioned thirteenth activity. Each pair sequentially represents plane one through six of the constellation. The first branch of the pair checks for three conditions to occur simultaneously. First, the plane must have failures in need of service. Second, one of the markers that track the location of available servicers (LL[21] through LL[26]) must hold that plane's number. Third, the servicer must not be in use at that moment. If the situation meets those three conditions, the following ASSIGN node increments the number of active servicers, and the ALTER node adds a unit of availability to the RESOURCE that corresponds to the plane in question. The second branch of the pair must also meet three simultaneous conditions. The conditions are the

same as the first branch of the pair except that the servicer must be in use. In the case of both branches of the pair, the entity continues on to the record sequence later in the second loop. If the state of the network does not satisfy any of these conditioned branches, it goes to the "Choose" QUEUE. The "ChooseP" SELECT node pulls the entity from the QUEUE and sends it through the next cyclically available emanating activity. The purpose of the cyclic selection rule from the SELECT node was to prevent the lower numbered planes from receiving service mission preference. Each activity was responsible for a different plane. The condition statement was had two main parts. First, the plane must have failures in need of service, and, second, none of the servicer location markers hold that plane's number. The following ASSIGN node then sets the first integer attribute for the entity (LTRIB[1]) to the number of the corresponding plane. The entity then enters the "Sort11" GOON node.

The "Sort11" GOON node begins the process of marking the location of the servicer or servicers. This node clones each entity that enters into two entities. If the entities carry an LTRIB[1] value that corresponds to a plane that the location markers already match, both entities are screened by the first two activities and are sent to close the GATE and terminate. If the markers do not already represent the plane of interest, the entities travel along the third and fourth activities from the "Sort11" node and update the markers. The "Split3" GOON node creates as many clones of the entity as there are servicers in the scenario. Then, if the number of servicers is greater than one, the values in the markers shift back a marker to make room for the number of the new plane that is using a servicer. The last activity following the "Split3" node sends the entity to the "Record" GOON node. The entity that goes along the fourth activity from the "Sort11"

node sets the first marker to the plane number that most recently received a servicer. The entity also increments the number of active servicers before going on to the "Needed" GOON node.

The "Record" node sends the entity along the first activity if the second activity is occupied. The purpose for the delay on the second activity is to ensure that all the updates to the variables are complete. The entity then triggers the WRITE node, which was used during development of the network.

The final portion of the second loop begins with the "Needed" GOON node. The delay on the emanating activity is the length of time it takes for a servicer to transfer from one plane to another. The condition on that activity is a check on the "Sort11" portion of the second loop. The "PChg1" ASSIGN node counts the number of plane changes that occur during the simulation. The ALTER node increments the units of availability for the appropriate RESOURCE.

Third Loop – Detailed Description

The responsibility of this loop is to collect data on the simulation and provide performance output. The "DETOUR3" ASSIGN node decrements the number of active satellites. The "1stServ" DETECT node releases an entity when the first service mission occurs, and the "1stServ2" COLLECT node records the time. The CREATE node generates four entities at a time when all satellites have experienced an unrepairable failure. It is important to check that all random failures have occurred at this create time. The first branch collects data on the time of the last service and the number of plane changes during that run of the simulation. The second branch calculates the average

extension of satellite life for satellites that warranted service. The second branch also calculates the mean repair time by the following equation.

$$\text{Mean Repair Time} = \frac{\text{Sum of Gap Times} + (3\ \text{months} * \text{Number of Missed Servicings})}{\text{Number of Servicing Missions} + \text{Number of Missed Servicings}}$$

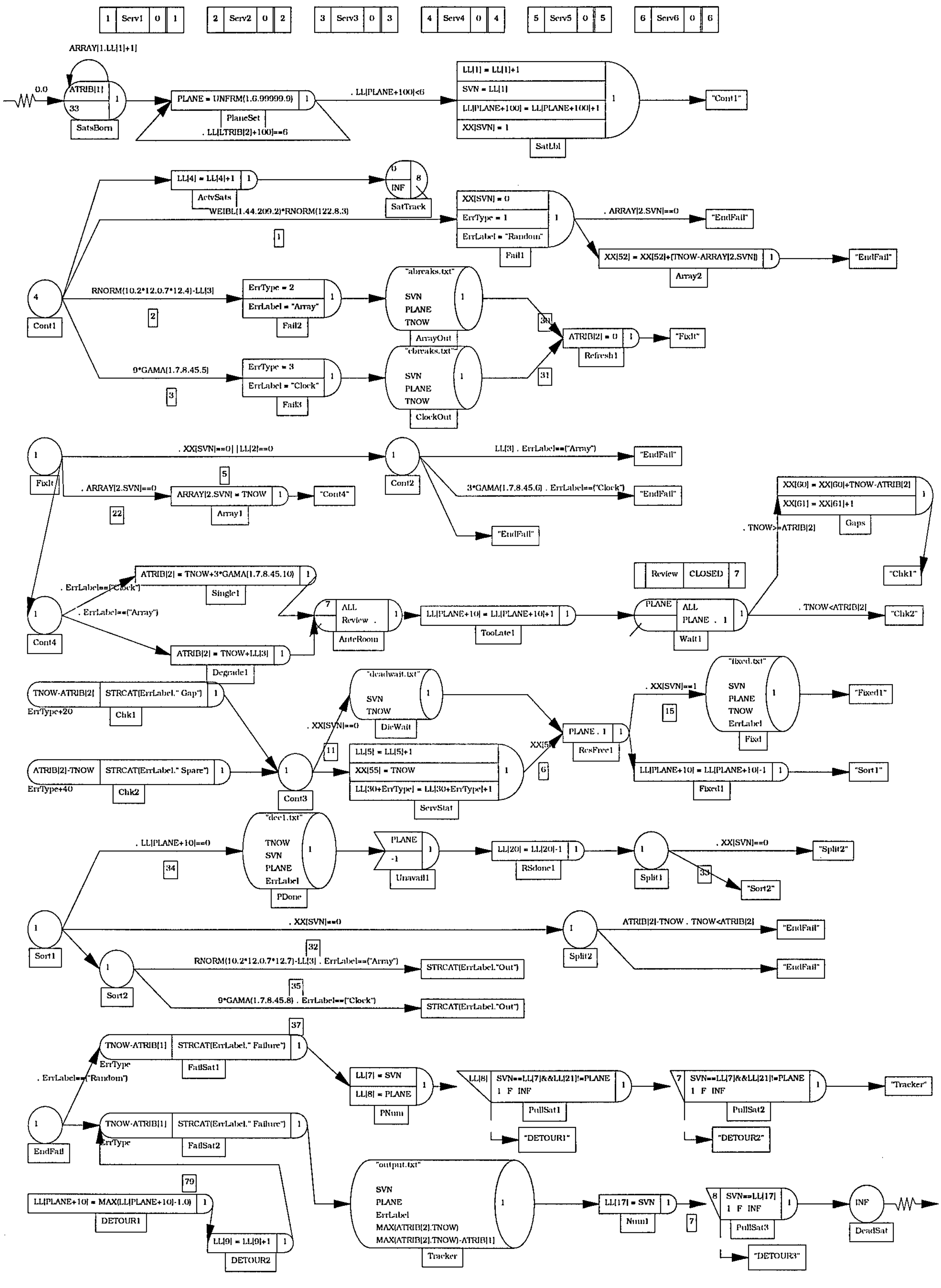
Equation 33

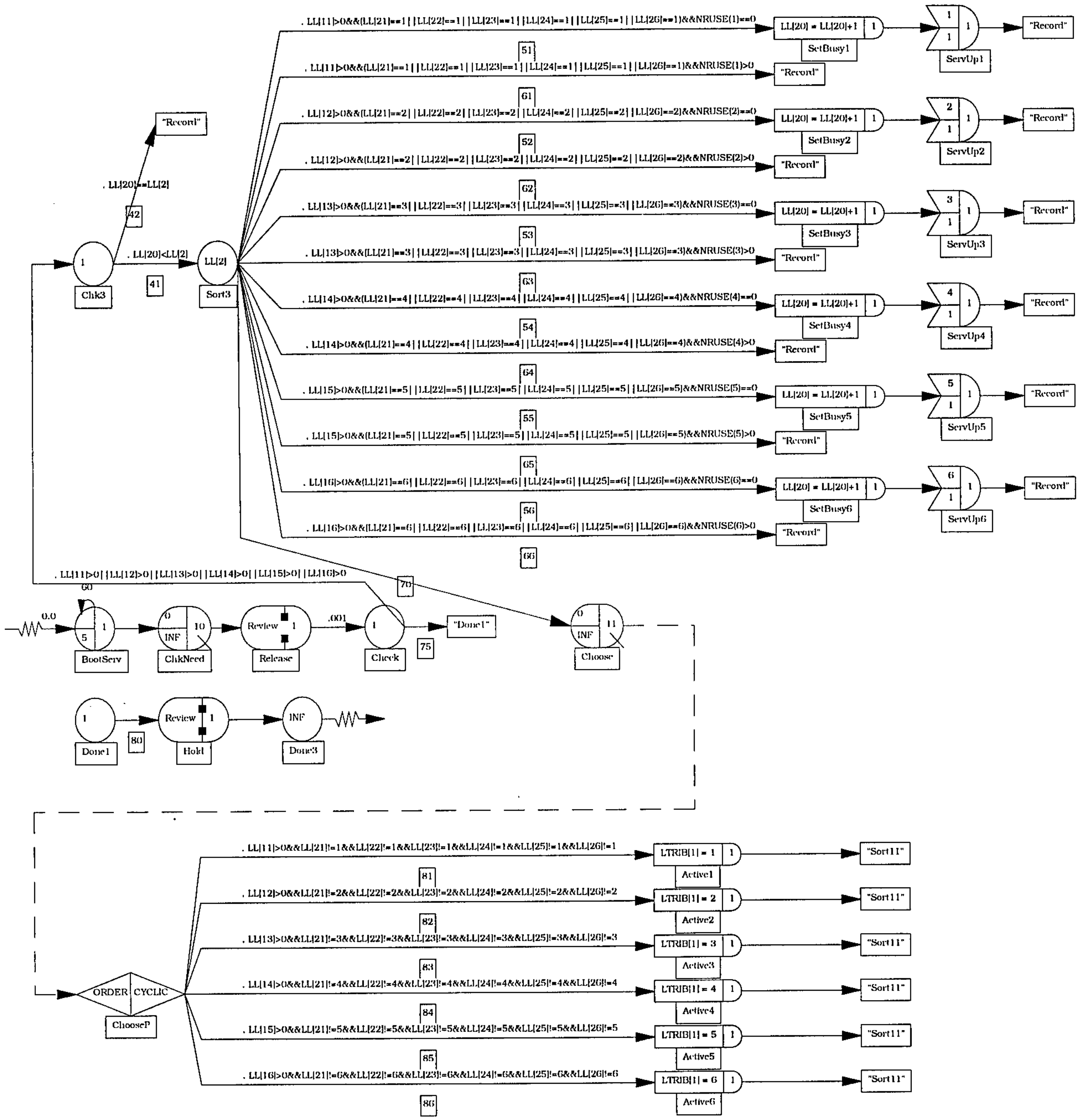
This essentially treats servicing repair times that occur prior to subsystem failure as zeros.

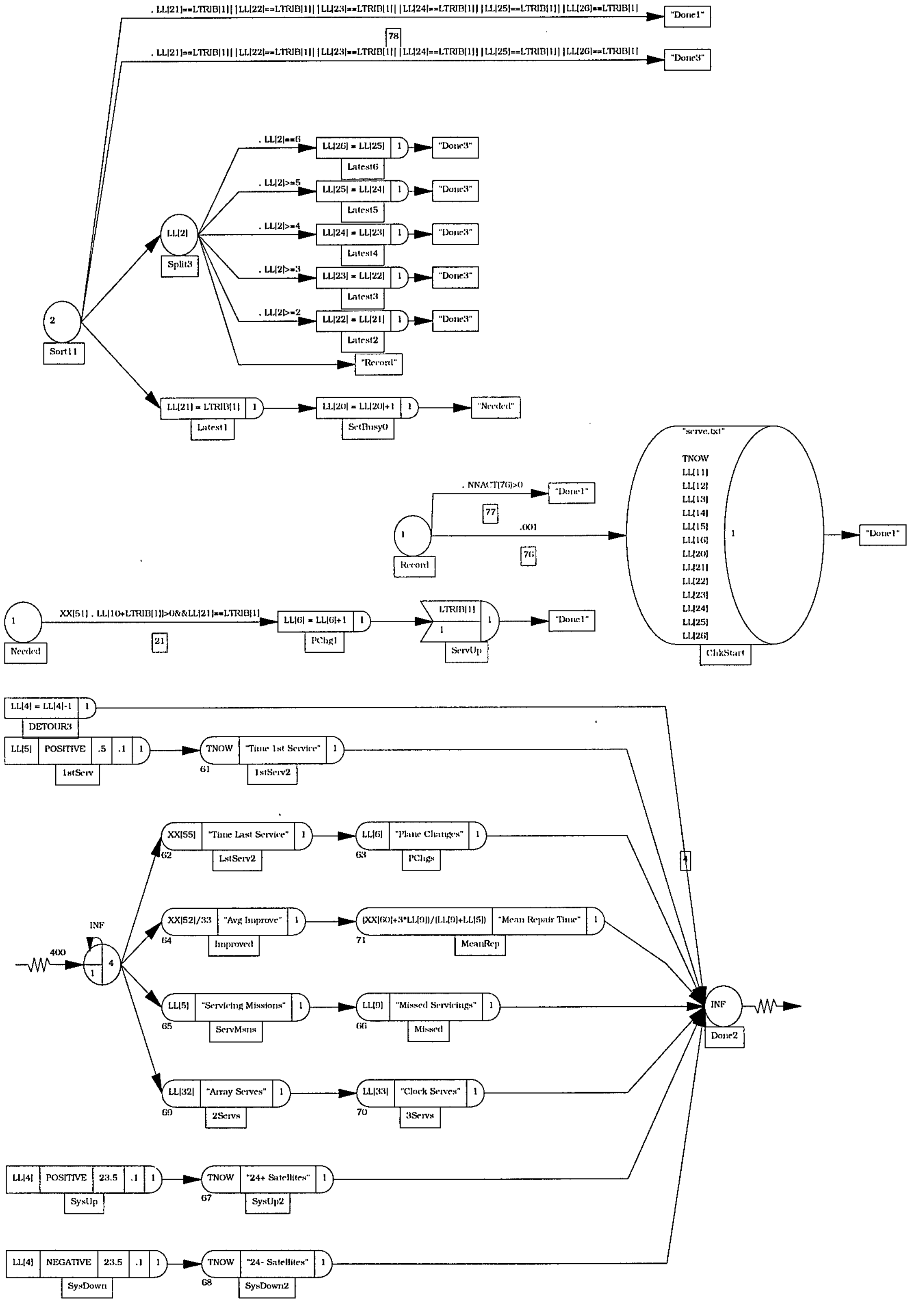
The third branch tracks the number of servicing missions and missed servicing opportunities for that run. The fourth branch tracks the number of servicing missions that occurred for each repairable subsystem. The "SysUp" DETECT node releases an entity when the constellation has 24 active satellites, and the following COLLECT node records the time. The "SysDown" DETECT node releases an entity when the number of satellites in the constellation falls below 24, and the following COLLECT node records the time.

Appendix T: AweSim Control File and Network for Architecture C

```
GEN,"Adam Wallen","Alternative C",18 Feb 99,20,YES,YES;
LIMITS,200,300,,10,10,10;
EQUIVALENCE,{{SVN,LTRIB[1]},{PLANE,LTRIB[2]},{ErrType,LTRIB[3]},{ErrLabel,STRIB
[1]}};
INTLC,{{LL[1],0},{LL[101],0},{LL[102],0},{LL[103],0},{LL[104],0},{LL[105],0},{L
L[106],0}};
INTLC,{{LL[121],0},{LL[122],0},{LL[123],0},{LL[124],0},{LL[125],0},{LL[126],0}}
;
INTLC,{{LL[2],6},{LL[3],3},{XX[50],20/30.5},{XX[51],0},{LL[9],0},{XX[60],0},{XX
[61],0}};
MONTR,RESOURCE(),TTBEG;
ARRAY,1,32,{6,4,4,4,6,6,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4};
ARRAY,2,33;
INITIALIZE,0.0,,YES,60;
NET;
FIN;
```


1 Serv1 0 1
2 Serv2 0 2
3 Serv3 0 3
4 Serv4 0 4
5 Serv5 0 5
6 Serv6 0 6
ARRAY[1,LL[1]+1]
0.0
ATRIB[1]
33
SatsBorn
PLANE = UNFRM(1.6,99999.9)
PlaneSet
. LL[LTRIB[2]+100]==6
. LL[PLANE+100]<6
LL[1] = LL[1]+1
SVN = LL[1]
LL[PLANE+100] = LL[PLANE+100]+1
XX[SVN] = 1
SatLbl
"Cont1"
LL[4] = LL[4]+1
ActvSats
WEIBL(1.44,209.2)*RNORM(122,8.3)
INF
SatTrack
XX[SVN] = 0
ErrType = 1
ErrLabel = "Random"
Fail1
. ARRAY[2,SVN]==0
"EndFail"
XX[52] = XX[52]+[TNOW-ARRAY[2,SVN]]
Array2
"EndFail"
Cont1
RNORM(10.2*12,0.7*12,4)-LL[3]
ErrType = 2
ErrLabel = "Array"
Fail2
"abreaks.txt"
SVN
PLANE
TNOW
ArrayOut
9*GAMA(1.7,8.45,5)
ErrType = 3
ErrLabel = "Clock"
Fail3
"cbreaks.txt"
ClockOut
ATRIB[2] = 0
Refresh1
"FixIt"
FixIt
. XX[SVN]==0| |LL[2]==0
. ARRAY[2,SVN]==0
ARRAY[2,SVN] = TNOW
Array1
"Cont4"
Cont2
LL[3] . ErrLabel==("Array")
"EndFail"
3*GAMA(1.7,8.45,6) . ErrLabel==("Clock")
"EndFail"
"EndFail"
XX[60] = XX[60]+TNOW-ATRIB[2]
XX[61] = XX[61]+1
Gaps
. TNOW>=ATRIB[2]
"Chk1"
Review CLOSED 7
. ErrLabel==("Clock")
ATRIB[2] = TNOW+3*GAMA(1.7,8.45,10)
Single1
. ErrLabel==("Array")
ALL
Review
AnteRoom
LL[PLANE+10] = LL[PLANE+10]+1
TooLate1
PLANE
ALL
PLANE
Wait1
. TNOW<ATRIB[2]
"Chk2"
Cont4
ATRIB[2] = TNOW+LL[3]
Degrade1
TNOW-ATRIB[2]
STRCAT(ErrLabel," Gap")
ErrType+20
Chk1
"deadwait.txt"
SVN
TNOW
DirWait
. XX[SVN]==0
PLANE
ResFree1
. XX[SVN]==1
"fixed.txt"
SVN
PLANE
TNOW
ErrLabel
Fixd
"Fixed1"
ATRIB[2]-TNOW
STRCAT(ErrLabel," Spare")
ErrType+40
Chk2
Cont3
LL[5] = LL[5]+1
XX[55] = TNOW
LL[30+ErrType] = LL[30+ErrType]+1
ServStat
LL[PLANE+10] = LL[PLANE+10]-1
Fixed1
"Sort1"
. LL[PLANE+10]==0
"decl.txt"
TNOW
SVN
PLANE
ErrLabel
PDone
PLANE
-1
Unavail1
LL[20] = LL[20]-1
RSdone1
Split1
. XX[SVN]==0
"Split2"
"Sort2"
Sort1
. XX[SVN]==0
Split2
ATRIB[2]-TNOW . TNOW<ATRIB[2]
"EndFail"
"EndFail"
Sort2
RNORM(10.2*12,0.7*12,7)-LL[3] . ErrLabel==("Array")
STRCAT(ErrLabel,"Out")
9*GAMA(1.7,8.45,8) . ErrLabel==("Clock")
STRCAT(ErrLabel,"Out")
TNOW-ATRIB[1]
STRCAT(ErrLabel," Failure")
ErrType
FailSat1
. ErrLabel==("Random")
LL[7] = SVN
LL[8] = PLANE
PNum
LL[8]
SVN==LL[7]&&LL[21]!=PLANE
1 F INF
PullSat1
"DETOUR1"
SVN==LL[7]&&LL[21]!=PLANE
1 F INF
PullSat2
"DETOUR2"
"Tracker"
EndFail
TNOW-ATRIB[1]
STRCAT(ErrLabel," Failure")
ErrType
FailSat2
LL[PLANE+10] = MAX(LL[PLANE+10]-1,0)
DETOUR1
LL[9] = LL[9]+1
DETOUR2
"output.txt"
SVN
PLANE
ErrLabel
MAX(ATRIB[2],TNOW)
MAX(ATRIB[2],TNOW)-ATRIB[1]
Tracker
LL[17] = SVN
Num1
SVN==LL[17]
1 F INF
PullSat3
"DETOUR3"
INF
DeadSat

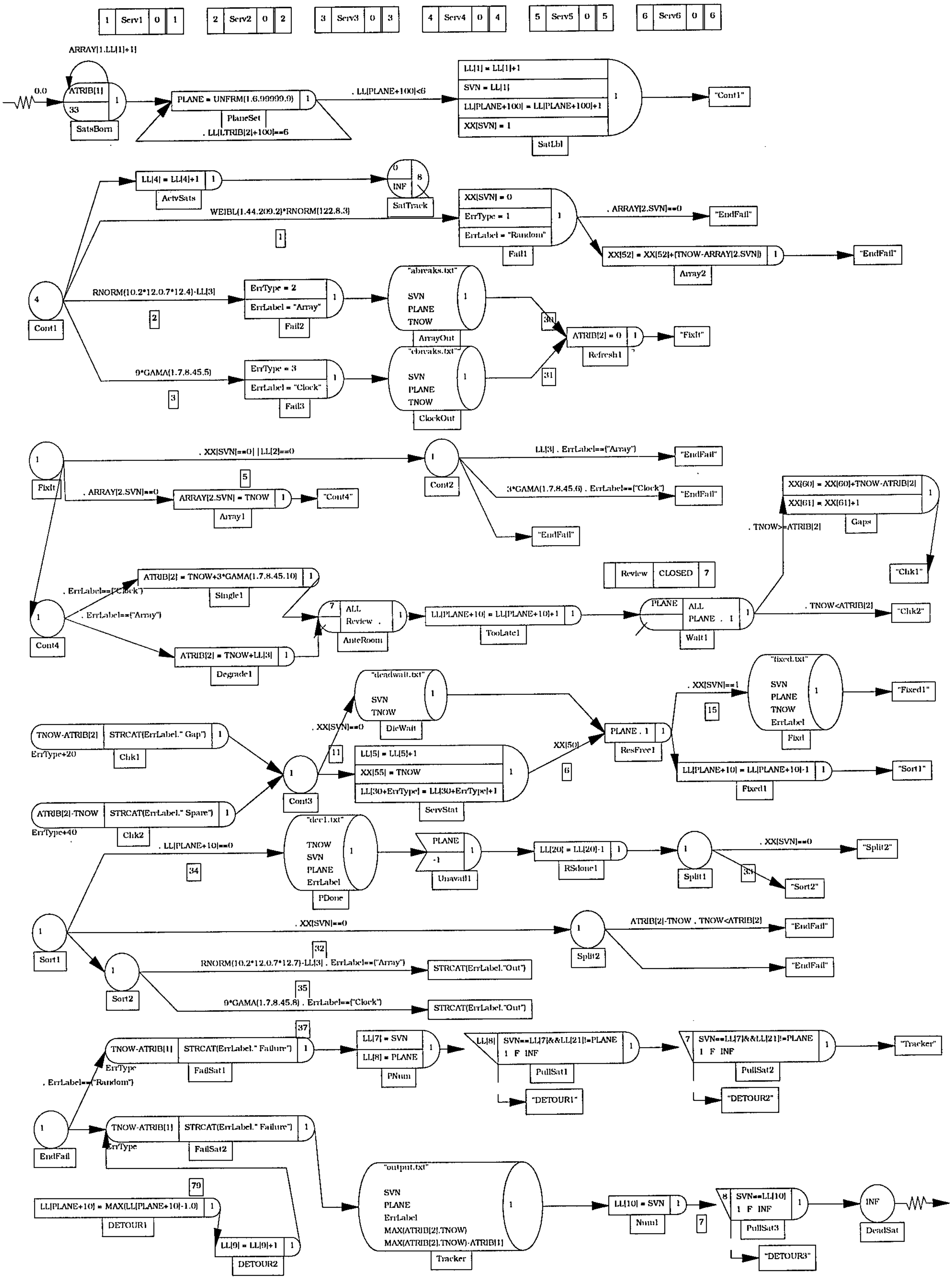

. LL[11]>0&&(LL[21]==1||LL[22]==1||LL[23]==1||LL[24]==1||LL[25]==1||LL[26]==1)&&NRUSE(1)==0
LL[20] = LL[20]+1
1
SetBusy1
51
ServUp1
"Record"
. LL[11]>0&&(LL[21]==1||LL[22]==1||LL[23]==1||LL[24]==1||LL[25]==1||LL[26]==1)&&NRUSE(1)>0
"Record"
61
. LL[12]>0&&(LL[21]==2||LL[22]==2||LL[23]==2||LL[24]==2||LL[25]==2||LL[26]==2)&&NRUSE(2)==0
SetBusy2
52
ServUp2
. LL[12]>0&&(LL[21]==2||LL[22]==2||LL[23]==2||LL[24]==2||LL[25]==2||LL[26]==2)&&NRUSE(2)>0
62
. LL[13]>0&&(LL[21]==3||LL[22]==3||LL[23]==3||LL[24]==3||LL[25]==3||LL[26]==3)&&NRUSE(3)==0
SetBusy3
53
ServUp3
. LL[13]>0&&(LL[21]==3||LL[22]==3||LL[23]==3||LL[24]==3||LL[25]==3||LL[26]==3)&&NRUSE(3)>0
63
. LL[14]>0&&(LL[21]==4||LL[22]==4||LL[23]==4||LL[24]==4||LL[25]==4||LL[26]==4)&&NRUSE(4)==0
SetBusy4
54
ServUp4
. LL[14]>0&&(LL[21]==4||LL[22]==4||LL[23]==4||LL[24]==4||LL[25]==4||LL[26]==4)&&NRUSE(4)>0
64
. LL[15]>0&&(LL[21]==5||LL[22]==5||LL[23]==5||LL[24]==5||LL[25]==5||LL[26]==5)&&NRUSE(5)==0
SetBusy5
55
ServUp5
. LL[15]>0&&(LL[21]==5||LL[22]==5||LL[23]==5||LL[24]==5||LL[25]==5||LL[26]==5)&&NRUSE(5)>0
65
. LL[16]>0&&(LL[21]==6||LL[22]==6||LL[23]==6||LL[24]==6||LL[25]==6||LL[26]==6)&&NRUSE(6)==0
SetBusy6
56
ServUp6
. LL[16]>0&&(LL[21]==6||LL[22]==6||LL[23]==6||LL[24]==6||LL[25]==6||LL[26]==6)&&NRUSE(6)>0
66
"Record"
. LL[20]>=LL[2]
42
Chk3
. LL[20]<LL[2]
41
LL[2]
Sort3
. LL[11]>0||LL[12]>0||LL[13]>0||LL[14]>0||LL[15]>0||LL[16]>0
60
76
0.0
5
BootServ
0
INF
10
ChkNeed
Review
Release
.001
Check
75
"Done1"
11
Choose
Done1
80
Hold
Done3
ORDER
CYCLIC
ChooseP
. LL[11]>0&&LL[21]!=1&&LL[22]!=1&&LL[23]!=1&&LL[24]!=1&&LL[25]!=1&&LL[26]!=1
LTRIB[1] = 1
Active1
81
"Sort11"
. LL[12]>0&&LL[21]!=2&&LL[22]!=2&&LL[23]!=2&&LL[24]!=2&&LL[25]!=2&&LL[26]!=2
LTRIB[1] = 2
Active2
82
. LL[13]>0&&LL[21]!=3&&LL[22]!=3&&LL[23]!=3&&LL[24]!=3&&LL[25]!=3&&LL[26]!=3
LTRIB[1] = 3
Active3
83
. LL[14]>0&&LL[21]!=4&&LL[22]!=4&&LL[23]!=4&&LL[24]!=4&&LL[25]!=4&&LL[26]!=4
LTRIB[1] = 4
Active4
84
. LL[15]>0&&LL[21]!=5&&LL[22]!=5&&LL[23]!=5&&LL[24]!=5&&LL[25]!=5&&LL[26]!=5
LTRIB[1] = 5
Active5
85
. LL[16]>0&&LL[21]!=6&&LL[22]!=6&&LL[23]!=6&&LL[24]!=6&&LL[25]!=6&&LL[26]!=6
LTRIB[1] = 6
Active6
86



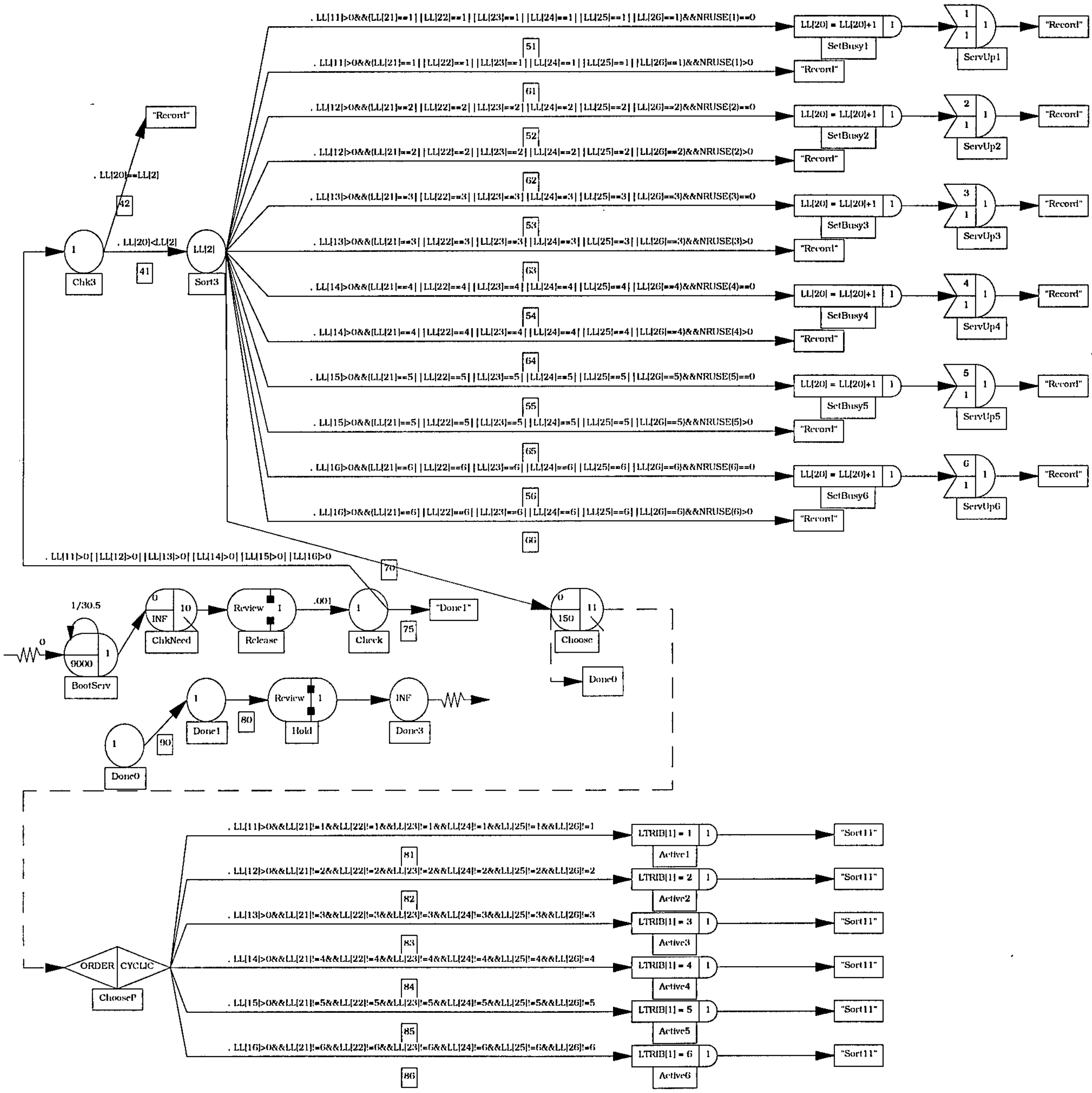
. LL[21]==LTRIB[1]||LL[22]==LTRIB[1]||LL[23]==LTRIB[1]||LL[24]==LTRIB[1]||LL[25]==LTRIB[1]||LL[26]==LTRIB[1]
"Done1"
78
. LL[21]==LTRIB[1]||LL[22]==LTRIB[1]||LL[23]==LTRIB[1]||LL[24]==LTRIB[1]||LL[25]==LTRIB[1]||LL[26]==LTRIB[1]
"Done3"
. LL[2]==6
LL[26] = LL[25]
1
"Done3"
Latest6
. LL[2]>=5
LL[25] = LL[24]
1
"Done3"
Latest5
LL[2]
Split3
. LL[2]>=4
LL[24] = LL[23]
1
"Done3"
Latest4
. LL[2]>=3
LL[23] = LL[22]
1
"Done3"
Latest3
2
Sort11
. LL[2]>=2
LL[22] = LL[21]
1
"Done3"
Latest2
"Record"
LL[21] = LTRIB[1]
1
Latest1
LL[20] = LL[20]+1
1
SetBusy0
"Needed"
"serve.txt"
TNOW
LL[11]
LL[12]
LL[13]
LL[14]
LL[15]
LL[16]
LL[20]
LL[21]
LL[22]
LL[23]
LL[24]
LL[25]
LL[26]
1
"Done1"
ClkStart
. NNACT[76]>0
"Done1"
77
1
.001
76
Record
1
XX[51] . LL[10+LTRIB[1]]>0&&LL[21]==LTRIB[1]
21
Needed
LL[6] = LL[6]+1
1
PChg1
LTRIB[1]
1
1
ServUp
"Done1"
LL[4] = LL[4]-1
1
DETOUR3
LL[5]
POSITIVE
.5
.1
1
1stServ
TNOW
"Time 1st Service"
1
61
1stServ2
XX[55]
"Time Last Service"
1
62
LstServ2
LL[6]
"Plane Changes"
1
63
PChgs
4
INF
400
4
1
XX[52]/33
"Avg Improve"
1
64
Improved
(XX[60]+3*LL[9])/(LL[9]+LL[5])
"Mean Repair Time"
1
71
MeanRep
LL[5]
"Servicing Missions"
1
65
ServMsns
LL[9]
"Missed Servicings"
1
66
Missed
INF
Done2
LL[32]
"Array Serves"
1
69
2Servs
LL[33]
"Clock Serves"
1
70
3Servs
LL[4]
POSITIVE
23.5
.1
1
SysUp
TNOW
"24+ Satellites"
1
67
SysUp2
LL[4]
NEGATIVE
23.5
.1
1
SysDown
TNOW
"24- Satellites"
1
68
SysDown2

Appendix U: AweSim Control File and Network for Architecture D

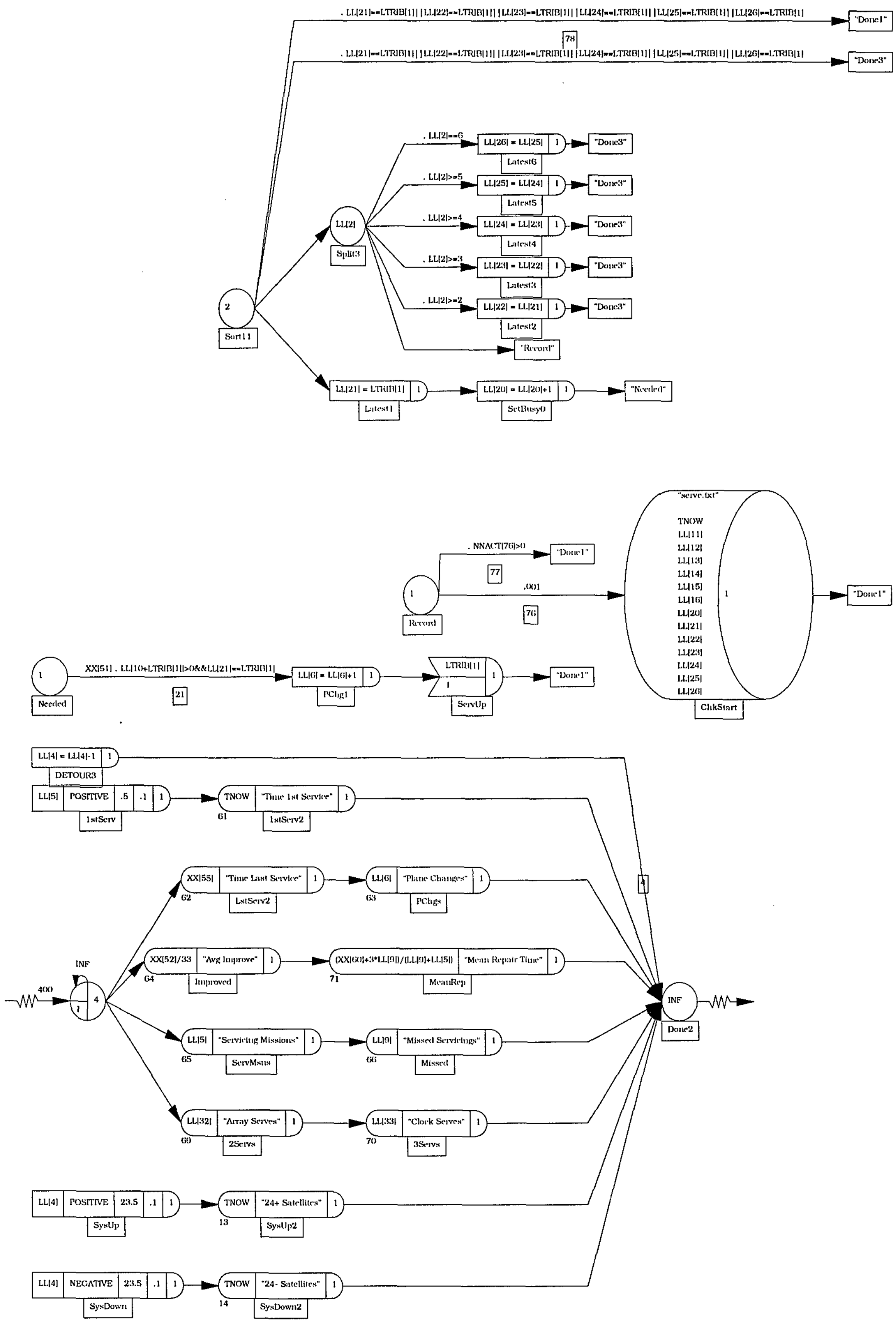
```
GEN,"Adam Wallen","Alternative D",18 Feb 99,20,YES,YES;
LIMITS,200,300,,10,10,10;
EQUIVALENCE,{{SVN,LTRIB[1]},{PLANE,LTRIB[2]},{ErrType,LTRIB[3]},{ErrLabel,STRIB
[1]}};
INTLC,{{LL[1],0},{LL[101],0},{LL[102],0},{LL[103],0},{LL[104],0},{LL[105],0},{L
L[106],0}};
INTLC,{{LL[121],0},{LL[122],0},{LL[123],0},{LL[124],0},{LL[125],0},{LL[126],0}}
;
INTLC,{{LL[2],6},{LL[3],3},{XX[50],12/30.5},{XX[51],0},{LL[9],0},{XX[60],0},{XX
[61],0}};
MONTR,RESOURCE(),TTBEG;
ARRAY,1,32,{6,4,4,4,6,6,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4};
ARRAY,2,33;
INITIALIZE,0.0,,YES,60;
NET;
FIN;
```



1 Serv1 0 1
2 Serv2 0 2
3 Serv3 0 3
4 Serv4 0 4
5 Serv5 0 5
6 Serv6 0 6
ARRAY[1,LL[1]+1]
0.0
ATRIB[1]
33
1
SatsBorn
PLANE = UNFRM(1,6.99999,0)
PlaneSet
, LL[LTRIB[2]+100]==6
, LL[PLANE+100]<6
LL[1] = LL[1]+1
SVN = LL[1]
LL[PLANE+100] = LL[PLANE+100]+1
XX[SVN] = 1
SatLbl
"Cont1"
LL[4] = LL[4]+1
ActvSats
0
8
INF
SatTrack
WEIBL(1,44,209.2)*RNORM(122,8,3)
XX[SVN] = 0
ErrType = 1
ErrLabel = "Random"
Fail1
, ARRAY[2,SVN]==0
"EndFail"
XX[52] = XX[52]+(TNOW-ARRAY[2,SVN])
Array2
"EndFail"
4
Cont1
RNORM(10.2*12,0.7*12,4)-LL[3]
2
ErrType = 2
ErrLabel = "Array"
Fail2
"abreaks.txt"
SVN
PLANE
TNOW
ArrayOut
30
ATRIB[2] = 0
Refresh1
"FixIt"
9*GAMA(1,7.8,45,5)
3
ErrType = 3
ErrLabel = "Clock"
Fail3
"cbreaks.txt"
SVN
PLANE
TNOW
ClockOut
31
FixIt
, XX[SVN]==0 | LL[2]==0
5
, ARRAY[2,SVN]==0
ARRAY[2,SVN] = TNOW
Array1
"Cont4"
Cont2
LL[3] , ErrLabel==("Array")
"EndFail"
3*GAMA(1,7.8,45,6) , ErrLabel==("Clock")
"EndFail"
"EndFail"
XX[60] = XX[60]+TNOW-ATRIB[2]
XX[61] = XX[61]+1
Gaps
, TNOW>=ATRIB[2]
"Chk1"
Review CLOSED 7
, ErrLabel==("Clock")
ATRIB[2] = TNOW+3*GAMA(1,7.8,45,10)
Single1
, ErrLabel==("Array")
Cont4
7 ALL Review
AnteRoom
LL[PLANE+10] = LL[PLANE+10]+1
TooLate1
PLANE
ALL
PLANE , 1
Wait1
, TNOW<ATRIB[2]
"Chk2"
ATRIB[2] = TNOW+LL[3]
Degrade1
"deadwait.txt"
SVN
TNOW
DieWait
, XX[SVN]==0
PLANE , 1
ResFree1
, XX[SVN]==1
15
"fixed.txt"
SVN
PLANE
TNOW
ErrLabel
Fixd
"Fixed1"
TNOW-ATRIB[2]
STRCAT(ErrLabel," Gap")
ErrType+20
Chk1
11
Cont3
LL[5] = LL[5]+1
XX[55] = TNOW
LL[30+ErrType] = LL[30+ErrType]+1
ServStat
XX[50]
6
LL[PLANE+10] = LL[PLANE+10]-1
Fixed1
"Sort1"
ATRIB[2]-TNOW
STRCAT(ErrLabel," Spare")
ErrType+40
Chk2
, LL[PLANE+10]==0
34
"dec1.txt"
TNOW
SVN
PLANE
ErrLabel
PDone
PLANE
-1
Unavail1
LL[20] = LL[20]-1
RSdone1
Split1
, XX[SVN]==0
"Split2"
33
"Sort2"
Sort1
, XX[SVN]==0
Split2
ATRIB[2]-TNOW , TNOW<ATRIB[2]
"EndFail"
"EndFail"
32
RNORM(10.2*12,0.7*12,7)-LL[3] , ErrLabel==("Array")
STRCAT(ErrLabel,"Out")
Sort2
35
9*GAMA(1,7.8,45,8) , ErrLabel==("Clock")
STRCAT(ErrLabel,"Out")
37
TNOW-ATRIB[1]
STRCAT(ErrLabel," Failure")
ErrType
FailSat1
, ErrLabel==("Random")
LL[7] = SVN
LL[8] = PLANE
PNum
LL[8]
SVN==LL[7]&&LL[21]!=PLANE
1 F INF
PullSat1
"DETOUR1"
7
SVN==LL[7]&&LL[21]!=PLANE
1 F INF
PullSat2
"DETOUR2"
"Tracker"
EndFail
TNOW-ATRIB[1]
STRCAT(ErrLabel," Failure")
ErrType
FailSat2
79
LL[PLANE+10] = MAX(LL[PLANE+10]-1,0)
DETOUR1
LL[9] = LL[9]+1
DETOUR2
"output.txt"
SVN
PLANE
ErrLabel
MAX(ATRIB[2],TNOW)
MAX(ATRIB[2],TNOW)-ATRIB[1]
Tracker
LL[10] = SVN
Num1
7
8
SVN==LL[10]
1 F INF
PullSat3
"DETOUR3"
INF
DeadSat



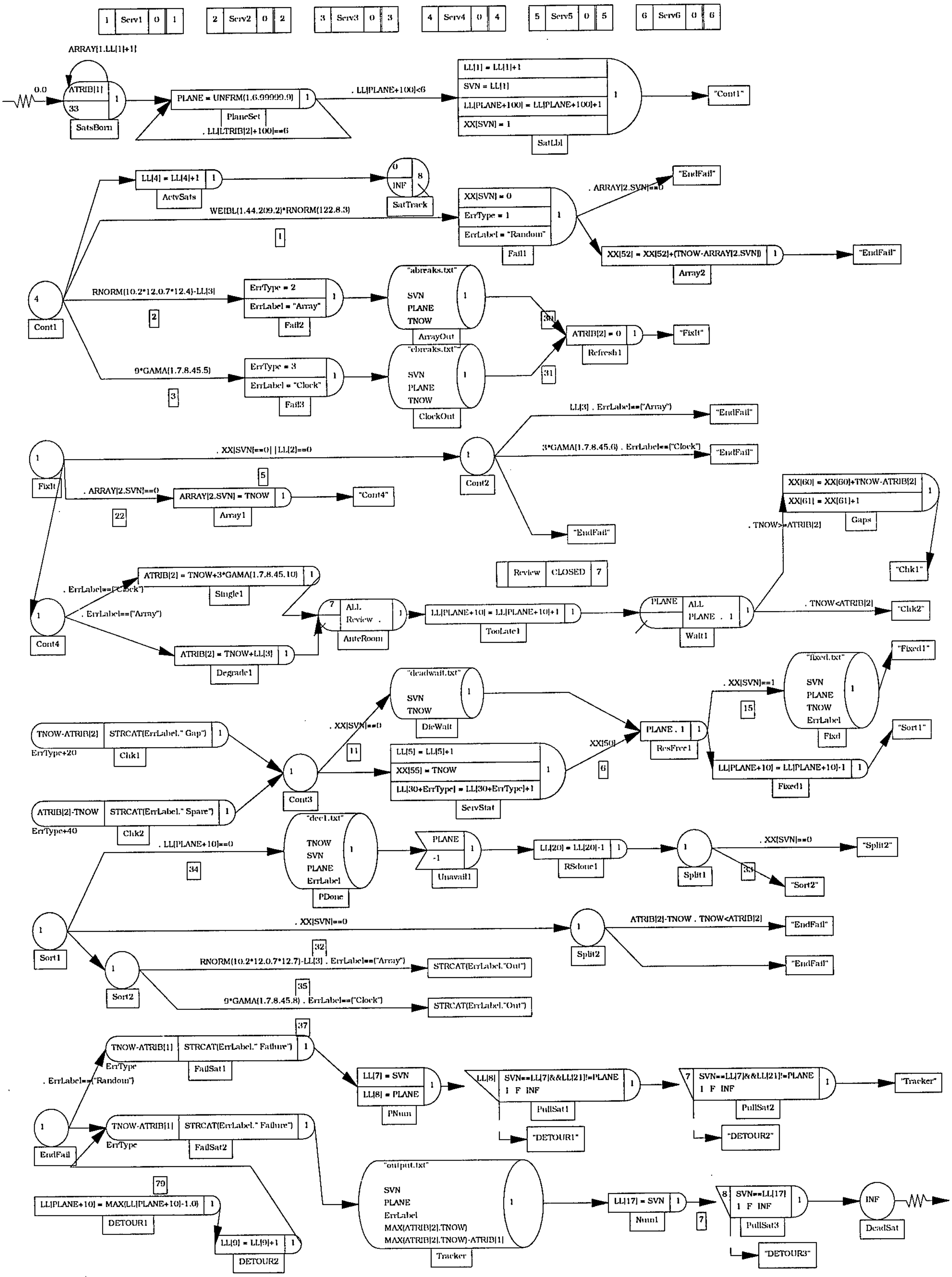
. LL[11]>0&&(LL[21]==1||LL[22]==1||LL[23]==1||LL[24]==1||LL[25]==1||LL[26]==1)&&NRUSE(1)==0
51
. LL[11]>0&&(LL[21]==1||LL[22]==1||LL[23]==1||LL[24]==1||LL[25]==1||LL[26]==1)&&NRUSE(1)>0
61
. LL[12]>0&&(LL[21]==2||LL[22]==2||LL[23]==2||LL[24]==2||LL[25]==2||LL[26]==2)&&NRUSE(2)==0
52
. LL[12]>0&&(LL[21]==2||LL[22]==2||LL[23]==2||LL[24]==2||LL[25]==2||LL[26]==2)&&NRUSE(2)>0
62
. LL[13]>0&&(LL[21]==3||LL[22]==3||LL[23]==3||LL[24]==3||LL[25]==3||LL[26]==3)&&NRUSE(3)==0
53
. LL[13]>0&&(LL[21]==3||LL[22]==3||LL[23]==3||LL[24]==3||LL[25]==3||LL[26]==3)&&NRUSE(3)>0
63
. LL[14]>0&&(LL[21]==4||LL[22]==4||LL[23]==4||LL[24]==4||LL[25]==4||LL[26]==4)&&NRUSE(4)==0
54
. LL[14]>0&&(LL[21]==4||LL[22]==4||LL[23]==4||LL[24]==4||LL[25]==4||LL[26]==4)&&NRUSE(4)>0
64
. LL[15]>0&&(LL[21]==5||LL[22]==5||LL[23]==5||LL[24]==5||LL[25]==5||LL[26]==5)&&NRUSE(5)==0
55
. LL[15]>0&&(LL[21]==5||LL[22]==5||LL[23]==5||LL[24]==5||LL[25]==5||LL[26]==5)&&NRUSE(5)>0
65
. LL[16]>0&&(LL[21]==6||LL[22]==6||LL[23]==6||LL[24]==6||LL[25]==6||LL[26]==6)&&NRUSE(6)==0
56
. LL[16]>0&&(LL[21]==6||LL[22]==6||LL[23]==6||LL[24]==6||LL[25]==6||LL[26]==6)&&NRUSE(6)>0
66
LL[20] = LL[20]+1
SetBusy1
SetBusy2
SetBusy3
SetBusy4
SetBusy5
SetBusy6
ServUp1
ServUp2
ServUp3
ServUp4
ServUp5
ServUp6
"Record"
. LL[20]==LL[2]
42
. LL[20]<LL[2]
41
Chk3
LL[2]
Sort3
. LL[11]>0||LL[12]>0||LL[13]>0||LL[14]>0||LL[15]>0||LL[16]>0
70
1/30.5
0
9000
BootServ
INF
10
ChkNeed
Review
Release
.001
Check
75
"Done1"
150
11
Choose
Done0
Done1
80
Hold
Done3
90
ORDER
CYCLIC
ChooseP
. LL[11]>0&&LL[21]!=1&&LL[22]!=1&&LL[23]!=1&&LL[24]!=1&&LL[25]!=1&&LL[26]!=1
81
. LL[12]>0&&LL[21]!=2&&LL[22]!=2&&LL[23]!=2&&LL[24]!=2&&LL[25]!=2&&LL[26]!=2
82
. LL[13]>0&&LL[21]!=3&&LL[22]!=3&&LL[23]!=3&&LL[24]!=3&&LL[25]!=3&&LL[26]!=3
83
. LL[14]>0&&LL[21]!=4&&LL[22]!=4&&LL[23]!=4&&LL[24]!=4&&LL[25]!=4&&LL[26]!=4
84
. LL[15]>0&&LL[21]!=5&&LL[22]!=5&&LL[23]!=5&&LL[24]!=5&&LL[25]!=5&&LL[26]!=5
85
. LL[16]>0&&LL[21]!=6&&LL[22]!=6&&LL[23]!=6&&LL[24]!=6&&LL[25]!=6&&LL[26]!=6
86
LTRIB[1] = 1
LTRIB[1] = 2
LTRIB[1] = 3
LTRIB[1] = 4
LTRIB[1] = 5
LTRIB[1] = 6
Active1
Active2
Active3
Active4
Active5
Active6
"Sort11"



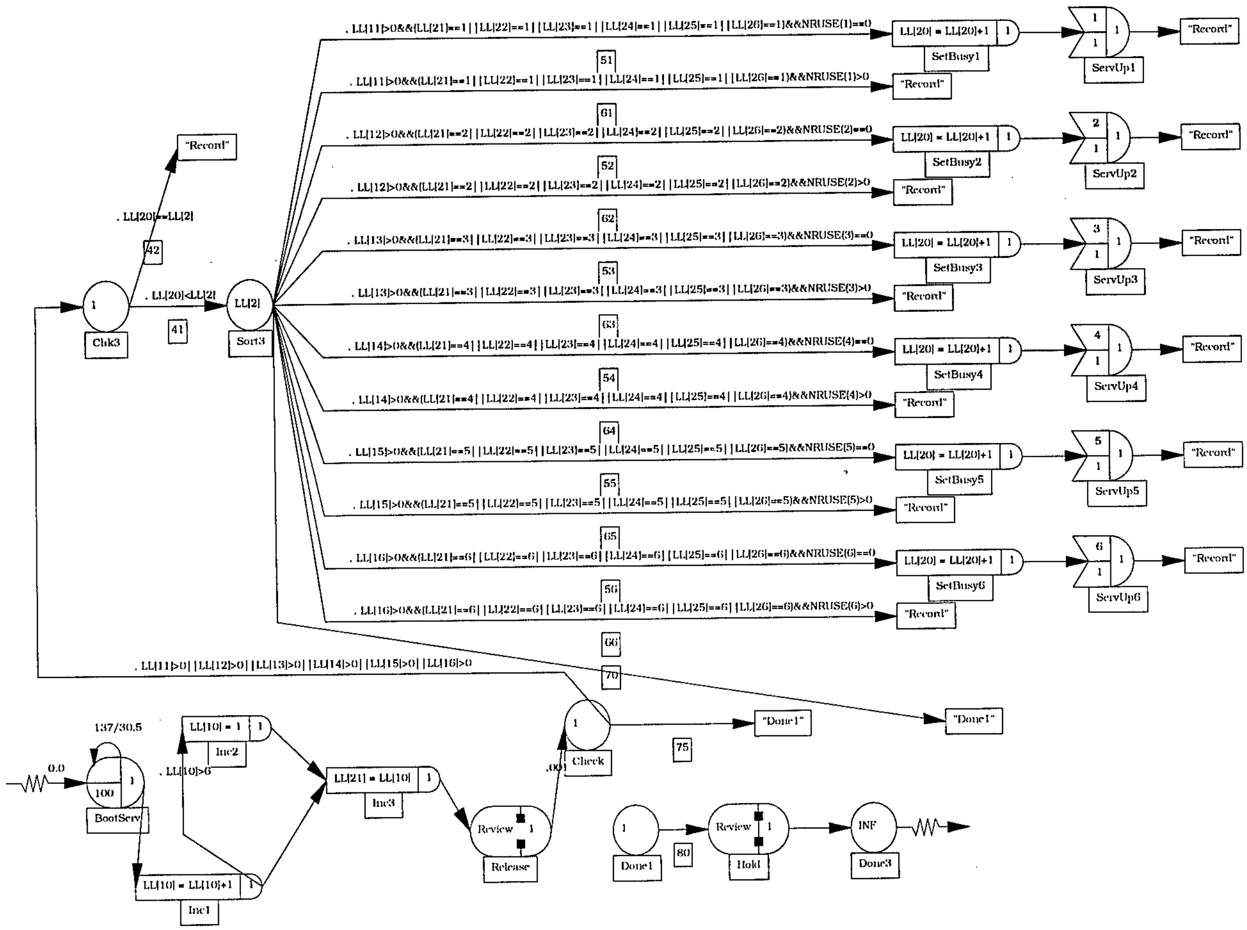
. LL[21]==LTRIB[1] | |LL[22]==LTRIB[1] | |LL[23]==LTRIB[1] | |LL[24]==LTRIB[1] | |LL[25]==LTRIB[1] | |LL[26]==LTRIB[1]
"Done1"
78
. LL[21]!=LTRIB[1] | |LL[22]!=LTRIB[1] | |LL[23]!=LTRIB[1] | |LL[24]!=LTRIB[1] | |LL[25]!=LTRIB[1] | |LL[26]!=LTRIB[1]
"Done3"
. LL[2]==6
LL[26] = LL[25]
1
"Done3"
Latest6
. LL[2]>=5
LL[25] = LL[24]
Latest5
. LL[2]>=4
LL[24] = LL[23]
Latest4
LL[2]
Split3
. LL[2]>=3
LL[23] = LL[22]
Latest3
. LL[2]>=2
LL[22] = LL[21]
Latest2
2
Sort11
"Record"
LL[21] = LTRIB[1]
Latest1
LL[20] = LL[20]+1
SetBusy0
"Needed"
"serve.txt"
TNOW
LL[11]
LL[12]
LL[13]
LL[14]
LL[15]
LL[16]
LL[20]
LL[21]
LL[22]
LL[23]
LL[24]
LL[25]
LL[26]
ClkStart
. NNACT[76]>0
"Done1"
77
.001
76
Record
XX[51] . LL[10+LTRIB[1]]>0&&LL[21]==LTRIB[1]
21
Needed
LL[6] = LL[6]+1
PChg1
LTRIB[1]
ServUp
LL[4] = LL[4]-1
DETOUR3
LL[5]
POSITIVE
.5
.1
1stServ
TNOW
"Time 1st Service"
61
1stServ2
4
XX[55]
"Time Last Service"
62
LstServ2
LL[6]
"Plane Changes"
63
PChgs
INF
400
XX[52]/33
"Avg Improve"
64
Improved
(XX[60]+3*LL[9])/(LL[9]+LL[5])
"Mean Repair Time"
71
MeanRep
INF
Done2
LL[5]
"Servicing Missions"
65
ServMsns
LL[9]
"Missed Servicings"
66
Missed
LL[32]
"Array Serves"
69
2Servs
LL[33]
"Clock Serves"
70
3Servs
LL[4]
POSITIVE
23.5
SysUp
TNOW
"24+ Satellites"
13
SysUp2
LL[4]
NEGATIVE
23.5
SysDown
TNOW
"24- Satellites"
14
SysDown2

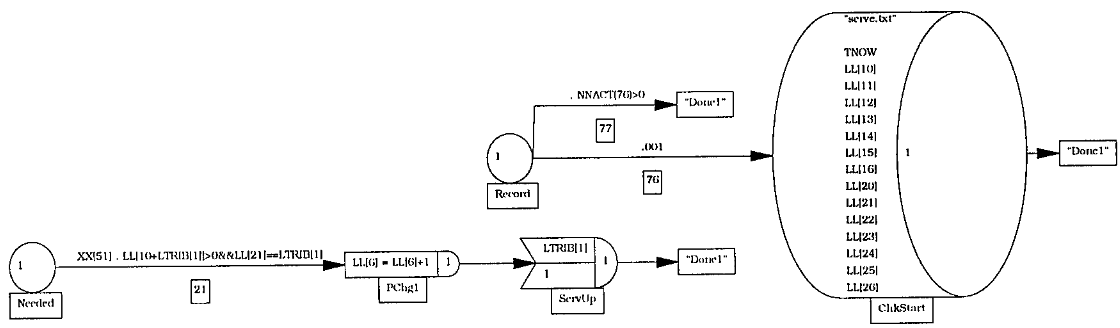
Appendix V: AweSim Control File and Network for Architecture E

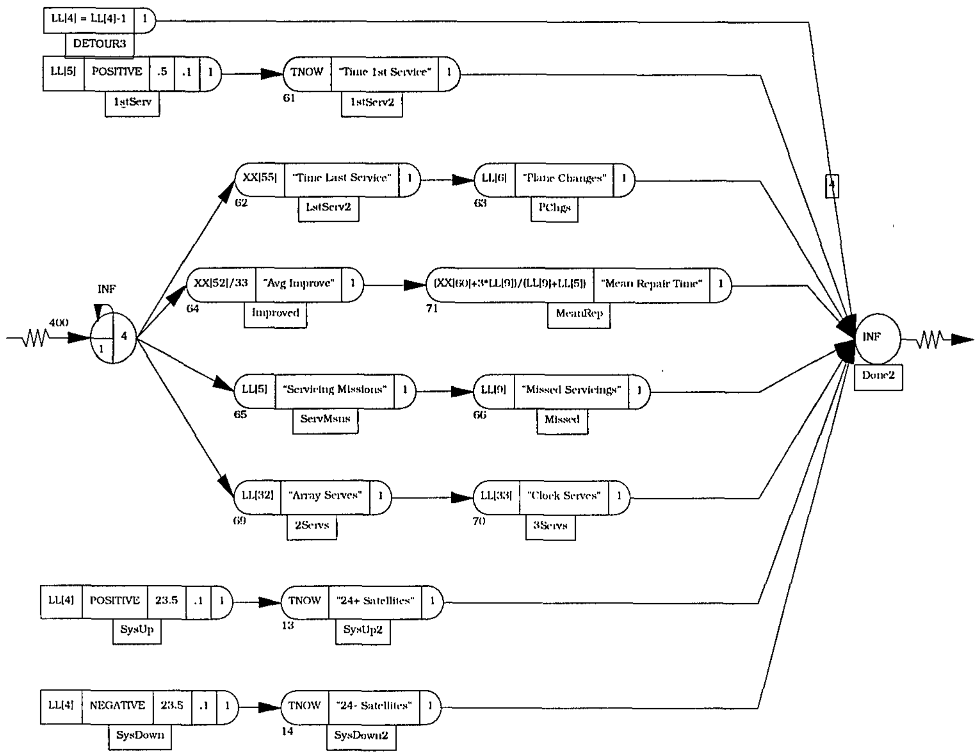
```
GEN,"Adam Wallen","Alternative E",18 Feb 99,20,YES,YES;
LIMITS,200,300,,10,10,10;
EQUIVALENCE,{{SVN,LTRIB[1]},{PLANE,LTRIB[2]},{ErrType,LTRIB[3]},{ErrLabel,STRIB
[1]}};
INTLC,{{LL[1],0},{LL[101],0},{LL[102],0},{LL[103],0},{LL[104],0},{LL[105],0},{L
L[106],0}};
INTLC,{{LL[121],0},{LL[122],0},{LL[123],0},{LL[124],0},{LL[125],0},{LL[126],0}}
;
INTLC,{{LL[2],1},{LL[3],3},{XX[50],8/30.5},{XX[51],0},{LL[9],0},{XX[60],0},{XX[
61],0}};
MONTR,RESOURCE(),TTBEG;
ARRAY,1,32,{6,4,4,4,6,6,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4};
ARRAY,2,33;
INITIALIZE,0.0,,YES,60;
NET;
FIN;
```



1 Serv1 0 1
2 Serv2 0 2
3 Serv3 0 3
4 Serv4 0 4
5 Serv5 0 5
6 Serv6 0 6
ARRAY[1,LL[1]+1]
0.0
ATRIB[1]
33
1
SatsBorn
PLANE = UNFRM(1,6.99999,9)
PlaneSet
. LL[LTRIB[2]+100]==6
. LL[PLANE+100]<6
LL[1] = LL[1]+1
SVN = LL[1]
LL[PLANE+100] = LL[PLANE+100]+1
XX[SVN] = 1
SatLbl
"Cont1"
LL[4] = LL[4]+1
ActvSats
0
INF
8
SatTrack
WEIBL(1.44,209.2)*RNORM(122,8,3)
1
XX[SVN] = 0
ErrType = 1
ErrLabel = "Random"
Fail1
. ARRAY[2,SVN]==0
"EndFail"
XX[52] = XX[52]+(TNOW-ARRAY[2,SVN])
Array2
"EndFail"
4
Cont1
RNORM(10.2*12,0.7*12,4)-LL[3]
2
ErrType = 2
ErrLabel = "Array"
Fail2
"abreaks.txt"
SVN
PLANE
TNOW
ArrayOut
30
ATRIB[2] = 0
Refresh1
"FixIt"
9*GAMA(1.7,8,45,5)
3
ErrType = 3
ErrLabel = "Clock"
Fail3
"cbreaks.txt"
SVN
PLANE
TNOW
ClockOut
31
LL[3] . ErrLabel==("Array")
"EndFail"
3*GAMA(1.7,8,45,6) . ErrLabel==("Clock")
"EndFail"
1
FixIt
. XX[SVN]==0 | LL[2]==0
5
Cont2
"EndFail"
. ARRAY[2,SVN]==0
22
ARRAY[2,SVN] = TNOW
Array1
"Cont4"
XX[60] = XX[60]+TNOW-ATRIB[2]
XX[61] = XX[61]+1
Gaps
. TNOW>=ATRIB[2]
"Chk1"
Review CLOSED 7
. ErrLabel==("Clock")
ATRIB[2] = TNOW+3*GAMA(1.7,8,45,10)
Single1
. ErrLabel==("Array")
Cont4
7 ALL
Review
AnteRoom
LL[PLANE+10] = LL[PLANE+10]+1
TooLate1
PLANE ALL
PLANE . 1
Wait1
. TNOW<ATRIB[2]
"Chk2"
ATRIB[2] = TNOW+LL[3]
Degrade1
"deadwait.txt"
SVN
TNOW
DieWait
. XX[SVN]==0
11
"fixed.txt"
SVN
PLANE
TNOW
ErrLabel
Fixd
"Fixed1"
. XX[SVN]==1
15
PLANE . 1
ResFree1
"Sort1"
TNOW-ATRIB[2]
STRCAT(ErrLabel," Gap")
ErrType+20
Chk1
Cont3
LL[5] = LL[5]+1
XX[55] = TNOW
LL[30+ErrType] = LL[30+ErrType]+1
ServStat
XX[50]
6
LL[PLANE+10] = LL[PLANE+10]-1
Fixed1
ATRIB[2]-TNOW
STRCAT(ErrLabel," Spare")
ErrType+40
Chk2
. LL[PLANE+10]==0
34
"decl.txt"
TNOW
SVN
PLANE
ErrLabel
PDone
PLANE
-1
Unavail1
LL[20] = LL[20]-1
RSdone1
Split1
. XX[SVN]==0
"Split2"
33
"Sort2"
Sort1
. XX[SVN]==0
ATRIB[2]-TNOW . TNOW<ATRIB[2]
"EndFail"
Split2
"EndFail"
32
RNORM(10.2*12,0.7*12,7)-LL[3] . ErrLabel==("Array")
STRCAT(ErrLabel,"Out")
Sort2
35
9*GAMA(1.7,8,45,8) . ErrLabel==("Clock")
STRCAT(ErrLabel,"Out")
37
TNOW-ATRIB[1]
STRCAT(ErrLabel," Failure")
ErrType
FailSat1
. ErrLabel==("Random")
LL[7] = SVN
LL[8] = PLANE
PNum
LL[8]
SVN==LL[7]&&LL[21]!=PLANE
1 F INF
PullSat1
"DETOUR1"
7
SVN==LL[7]&&LL[21]!=PLANE
1 F INF
PullSat2
"DETOUR2"
"Tracker"
EndFail
TNOW-ATRIB[1]
STRCAT(ErrLabel," Failure")
ErrType
FailSat2
79
LL[PLANE+10] = MAX(LL[PLANE+10]-1,0)
DETOUR1
LL[9] = LL[9]+1
DETOUR2
"output.txt"
SVN
PLANE
ErrLabel
MAX(ATRIB[2],TNOW)
MAX(ATRIB[2],TNOW)-ATRIB[1]
Tracker
LL[17] = SVN
Num1
7
8
SVN==LL[17]
1 F INF
PullSat3
"DETOUR3"
INF
DeadSat


. LL[11]>0&&(LL[21]==1||LL[22]==1||LL[23]==1||LL[24]==1||LL[25]==1||LL[26]==1)&&NRUSE(1)==0
LL[20] = LL[20]+1
SetBusy1
ServUp1
"Record"
51
. LL[11]>0&&(LL[21]==1||LL[22]==1||LL[23]==1||LL[24]==1||LL[25]==1||LL[26]==1)&&NRUSE(1)>0
61
. LL[12]>0&&(LL[21]==2||LL[22]==2||LL[23]==2||LL[24]==2||LL[25]==2||LL[26]==2)&&NRUSE(2)==0
SetBusy2
ServUp2
52
. LL[12]>0&&(LL[21]==2||LL[22]==2||LL[23]==2||LL[24]==2||LL[25]==2||LL[26]==2)&&NRUSE(2)>0
62
. LL[13]>0&&(LL[21]==3||LL[22]==3||LL[23]==3||LL[24]==3||LL[25]==3||LL[26]==3)&&NRUSE(3)==0
SetBusy3
ServUp3
53
. LL[13]>0&&(LL[21]==3||LL[22]==3||LL[23]==3||LL[24]==3||LL[25]==3||LL[26]==3)&&NRUSE(3)>0
63
. LL[14]>0&&(LL[21]==4||LL[22]==4||LL[23]==4||LL[24]==4||LL[25]==4||LL[26]==4)&&NRUSE(4)==0
SetBusy4
ServUp4
54
. LL[14]>0&&(LL[21]==4||LL[22]==4||LL[23]==4||LL[24]==4||LL[25]==4||LL[26]==4)&&NRUSE(4)>0
64
. LL[15]>0&&(LL[21]==5||LL[22]==5||LL[23]==5||LL[24]==5||LL[25]==5||LL[26]==5)&&NRUSE(5)==0
SetBusy5
ServUp5
55
. LL[15]>0&&(LL[21]==5||LL[22]==5||LL[23]==5||LL[24]==5||LL[25]==5||LL[26]==5)&&NRUSE(5)>0
65
. LL[16]>0&&(LL[21]==6||LL[22]==6||LL[23]==6||LL[24]==6||LL[25]==6||LL[26]==6)&&NRUSE(6)==0
SetBusy6
ServUp6
56
. LL[16]>0&&(LL[21]==6||LL[22]==6||LL[23]==6||LL[24]==6||LL[25]==6||LL[26]==6)&&NRUSE(6)>0
66
70
. LL[20]==LL[2]
42
. LL[20]<LL[2]
41
Chk3
LL[2]
Sort3
. LL[11]>0||LL[12]>0||LL[13]>0||LL[14]>0||LL[15]>0||LL[16]>0
"Done1"
75
137/30.5
0.0
100
BootServ
LL[10] = 1
Inc2
. LL[10]>6
LL[10] = LL[10]+1
Inc1
LL[21] = LL[10]
Inc3
Review
Release
.001
Check
Done1
80
Hold
INF
Done3

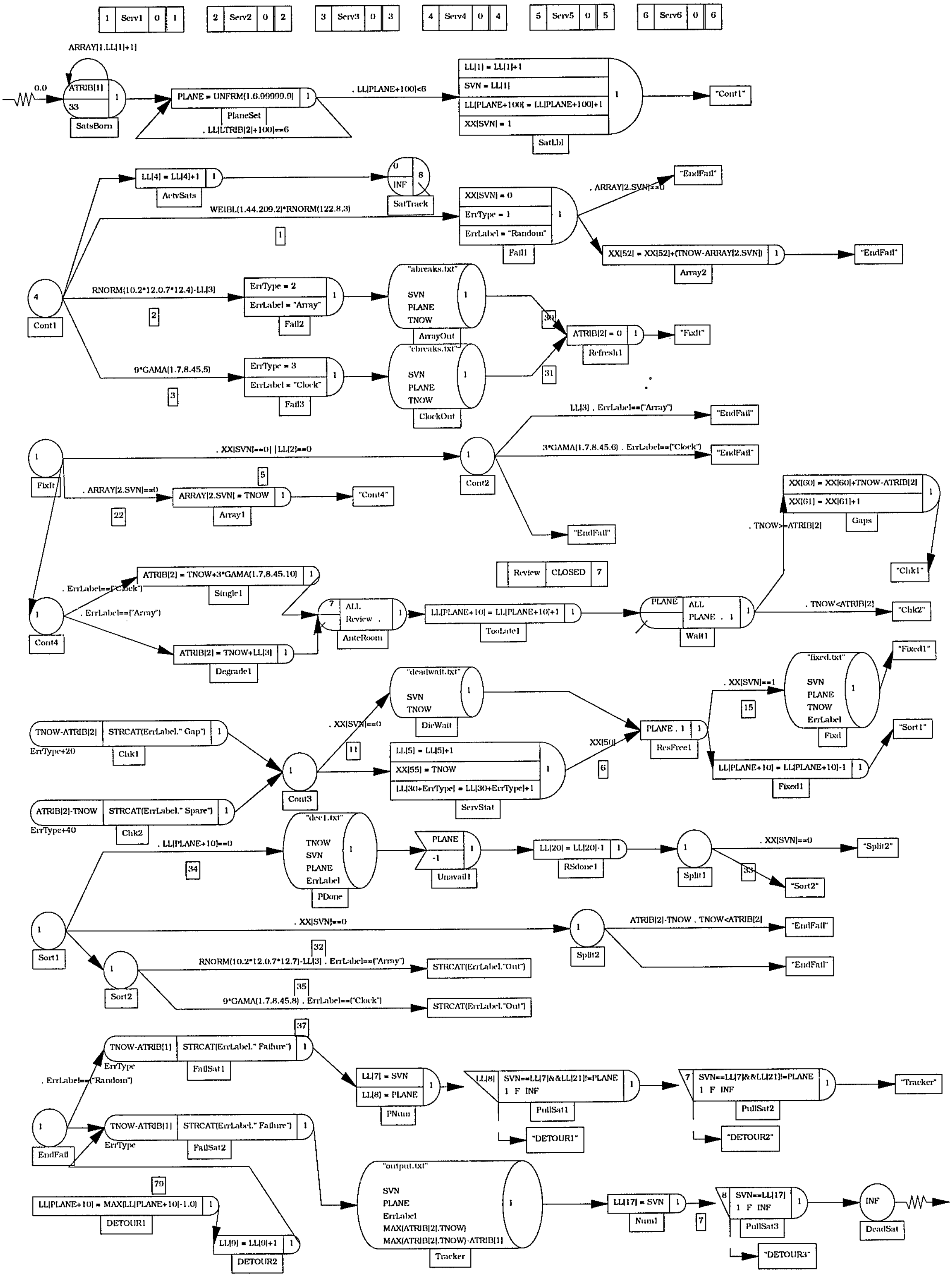

. NNACT(76)>0
"Done1"
77
.001
76
Record
"serve.txt"
TNOW
LL[10]
LL[11]
LL[12]
LL[13]
LL[14]
LL[15]
LL[16]
LL[20]
LL[21]
LL[22]
LL[23]
LL[24]
LL[25]
LL[26]
ChkStart
XX[51] . LL[10+LTRIB[1]]>0&&LL[21]==LTRIB[1]
21
Needed
LL[6] = LL[6]+1
PChg1
LTRIB[1]
ServUp



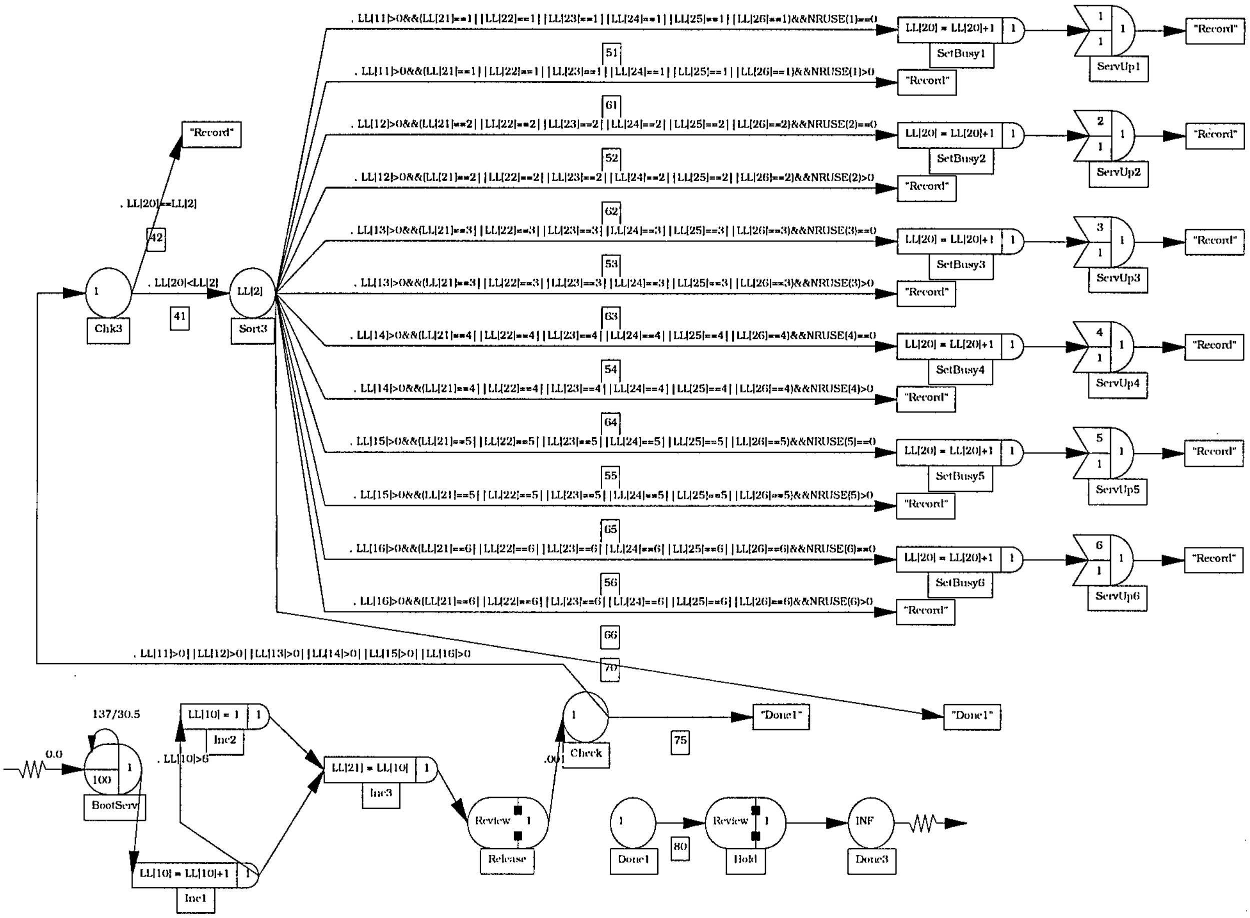
LL[4] = LL[4]-1
1
DETOUR3
LL[5]
POSITIVE
.5
.1
1
1stServ
TNOW
"Time 1st Service"
1
61
1stServ2
XX[55]
"Time Last Service"
1
62
LstServ2
LL[6]
"Plane Changes"
1
63
PChgs
INF
400
1
4
XX[52]/33
"Avg Improve"
1
64
Improved
(XX[60]+3*LL[9])/(LL[9]+LL[5])
"Mean Repair Time"
1
71
MeanRep
LL[5]
"Servicing Missions"
1
65
ServMsns
LL[9]
"Missed Servicings"
1
66
Missed
LL[32]
"Array Serves"
1
69
2Servs
LL[33]
"Clock Serves"
1
70
3Servs
LL[4]
POSITIVE
23.5
.1
1
SysUp
TNOW
"24+ Satellites"
1
13
SysUp2
LL[4]
NEGATIVE
23.5
.1
1
SysDown
TNOW
"24- Satellites"
1
14
SysDown2
INF
Done2

Appendix W: AweSim Control File and Network for Architecture F

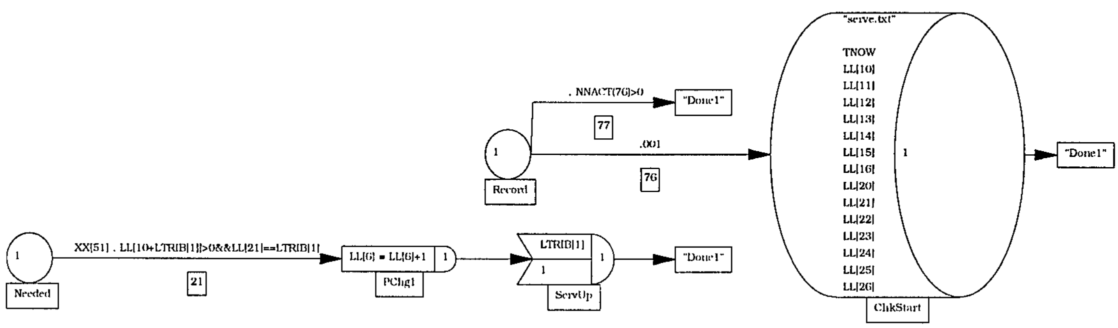
```
GEN,"Adam Wallen","Alternative F",18 Feb 99,20,YES,YES;
LIMITS,200,300,,10,10,10;
EQUIVALENCE,{{SVN,LTRIB[1]},{PLANE,LTRIB[2]},{ErrType,LTRIB[3]},{ErrLabel,STRIB
[1]}};
INTLC,{{LL[1],0},{LL[101],0},{LL[102],0},{LL[103],0},{LL[104],0},{LL[105],0},{L
L[106],0}};
INTLC,{{LL[121],0},{LL[122],0},{LL[123],0},{LL[124],0},{LL[125],0},{LL[126],0}}
;
INTLC,{{LL[2],1},{LL[3],3},{XX[50],4/30.5},{XX[51],0},{LL[9],0}};
MONTR,RESOURCE(),TTBEG;
ARRAY,1,32,{6,4,4,4,6,6,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4,4};
ARRAY,2,33;
INITIALIZE,0.0,,YES,60;
NET;
FIN;
```



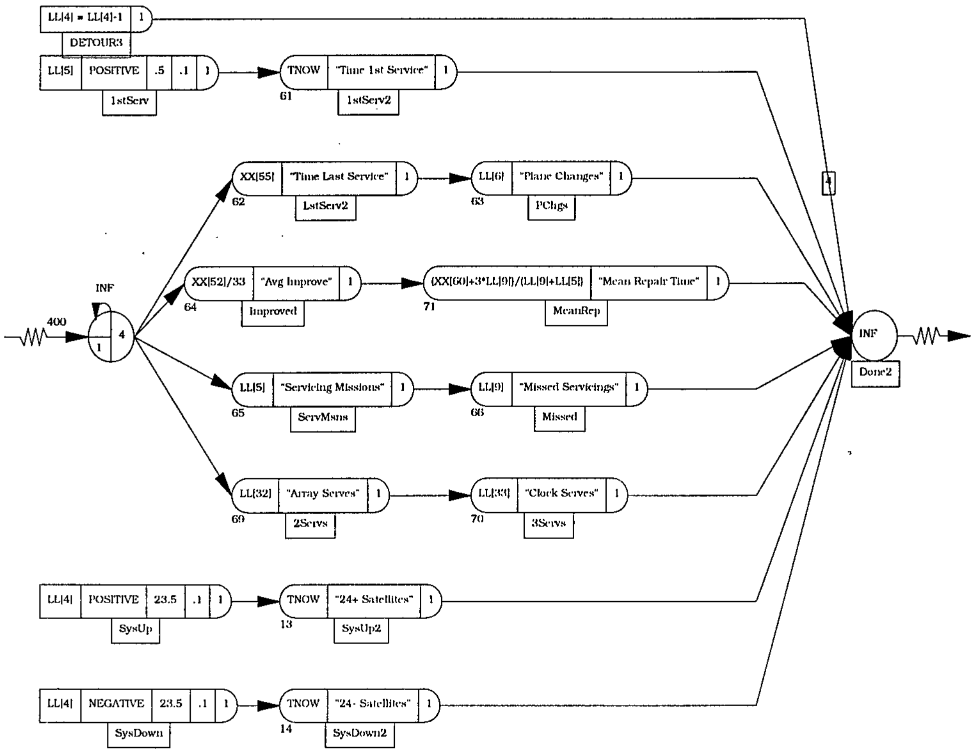
1 Serv1 0 1
2 Serv2 0 2
3 Serv3 0 3
4 Serv4 0 4
5 Serv5 0 5
6 Serv6 0 6
ARRAY[1,LL[1]+1]
0.0
ATRIB[1]
33
SatsBorn
PLANE = UNFRM(1,6.99999,9)
PlaneSet
. LL[LTRIB[2]+100]==6
. LL[PLANE+100]<6
LL[1] = LL[1]+1
SVN = LL[1]
LL[PLANE+100] = LL[PLANE+100]+1
XX[SVN] = 1
SatLbl
"Cont1"
LL[4] = LL[4]+1
ActvSats
SatTrack
INF
WEIBL(1.44,209.2)*RNORM(122.8,3)
XX[SVN] = 0
ErrType = 1
ErrLabel = "Random"
Fail1
. ARRAY[2,SVN]==0
"EndFail"
XX[52] = XX[52]+(TNOW-ARRAY[2,SVN])
Array2
Cont1
RNORM(10.2*12,0.7*12.4)-LL[3]
ErrType = 2
ErrLabel = "Array"
Fail2
"abreaks.txt"
SVN
PLANE
TNOW
ArrayOut
ATRIB[2] = 0
Refresh1
"FixIt"
9*GAMA(1.7,8.45,5)
ErrType = 3
ErrLabel = "Clock"
Fail3
"cbreaks.txt"
ClockOut
LL[3] . ErrLabel==("Array")
3*GAMA(1.7,8.45,6) . ErrLabel==("Clock")
FixIt
. XX[SVN]==0| |LL[2]==0
Cont2
. ARRAY[2,SVN]==0
ARRAY[2,SVN] = TNOW
Array1
"Cont4"
XX[60] = XX[60]+TNOW-ATRIB[2]
XX[61] = XX[61]+1
Gaps
. TNOW>=ATRIB[2]
"Chk1"
Review CLOSED 7
. ErrLabel==("Clock")
ATRIB[2] = TNOW+3*GAMA(1.7,8.45,10)
Single1
. ErrLabel==("Array")
Cont4
ALL
Review
AnteRoom
LL[PLANE+10] = LL[PLANE+10]+1
Toolate1
PLANE
ALL
PLANE . 1
Wait1
. TNOW<ATRIB[2]
"Chk2"
ATRIB[2] = TNOW+LL[3]
Degrade1
"deadwait.txt"
SVN
TNOW
DieWait
. XX[SVN]==1
"fixed.txt"
SVN
PLANE
TNOW
ErrLabel
Fixd
"Fixed1"
TNOW-ATRIB[2]
STRCAT(ErrLabel," Gap")
ErrType+20
Chk1
. XX[SVN]==0
Cont3
LL[5] = LL[5]+1
XX[55] = TNOW
LL[30+ErrType] = LL[30+ErrType]+1
ServStat
XX[50]
PLANE . 1
ResFree1
LL[PLANE+10] = LL[PLANE+10]-1
Fixed1
"Sort1"
ATRIB[2]-TNOW
STRCAT(ErrLabel," Spare")
ErrType+40
Chk2
. LL[PLANE+10]==0
"dec1.txt"
TNOW
SVN
PLANE
ErrLabel
PDone
PLANE
-1
Unavail1
LL[20] = LL[20]-1
RSdone1
Split1
. XX[SVN]==0
"Split2"
"Sort2"
Sort1
. XX[SVN]==0
Split2
ATRIB[2]-TNOW . TNOW<ATRIB[2]
"EndFail"
Sort2
RNORM(10.2*12,0.7*12.7)-LL[3] . ErrLabel==("Array")
STRCAT(ErrLabel,"Out")
9*GAMA(1.7,8.45,8) . ErrLabel==("Clock")
TNOW-ATRIB[1]
STRCAT(ErrLabel," Failure")
ErrType
FailSat1
. ErrLabel==("Random")
LL[7] = SVN
LL[8] = PLANE
PNum
LL[8]
SVN==LL[7]&&LL[21]!=PLANE
1 F INF
PullSat1
"DETOUR1"
SVN==LL[7]&&LL[21]!=PLANE
1 F INF
PullSat2
"DETOUR2"
"Tracker"
EndFail
FailSat2
"output.txt"
SVN
PLANE
ErrLabel
MAX(ATRIB[2],TNOW)
MAX(ATRIB[2],TNOW)-ATRIB[1]
Tracker
LL[17] = SVN
Num1
SVN==LL[17]
1 F INF
PullSat3
"DETOUR3"
INF
DeadSat
LL[PLANE+10] = MAX(LL[PLANE+10]-1,0)
DETOUR1
LL[9] = LL[9]+1
DETOUR2



. LL[11]>0&&(LL[21]==1||LL[22]==1||LL[23]==1||LL[24]==1||LL[25]==1||LL[26]==1)&&NRUSE(1)==0
51
. LL[11]>0&&(LL[21]==1||LL[22]==1||LL[23]==1||LL[24]==1||LL[25]==1||LL[26]==1)&&NRUSE(1)>0
61
. LL[12]>0&&(LL[21]==2||LL[22]==2||LL[23]==2||LL[24]==2||LL[25]==2||LL[26]==2)&&NRUSE(2)==0
52
. LL[12]>0&&(LL[21]==2||LL[22]==2||LL[23]==2||LL[24]==2||LL[25]==2||LL[26]==2)&&NRUSE(2)>0
62
. LL[13]>0&&(LL[21]==3||LL[22]==3||LL[23]==3||LL[24]==3||LL[25]==3||LL[26]==3)&&NRUSE(3)==0
53
. LL[13]>0&&(LL[21]==3||LL[22]==3||LL[23]==3||LL[24]==3||LL[25]==3||LL[26]==3)&&NRUSE(3)>0
63
. LL[14]>0&&(LL[21]==4||LL[22]==4||LL[23]==4||LL[24]==4||LL[25]==4||LL[26]==4)&&NRUSE(4)==0
54
. LL[14]>0&&(LL[21]==4||LL[22]==4||LL[23]==4||LL[24]==4||LL[25]==4||LL[26]==4)&&NRUSE(4)>0
64
. LL[15]>0&&(LL[21]==5||LL[22]==5||LL[23]==5||LL[24]==5||LL[25]==5||LL[26]==5)&&NRUSE(5)==0
55
. LL[15]>0&&(LL[21]==5||LL[22]==5||LL[23]==5||LL[24]==5||LL[25]==5||LL[26]==5)&&NRUSE(5)>0
65
. LL[16]>0&&(LL[21]==6||LL[22]==6||LL[23]==6||LL[24]==6||LL[25]==6||LL[26]==6)&&NRUSE(6)==0
56
. LL[16]>0&&(LL[21]==6||LL[22]==6||LL[23]==6||LL[24]==6||LL[25]==6||LL[26]==6)&&NRUSE(6)>0
66
LL[20] = LL[20]+1
SetBusy1
SetBusy2
SetBusy3
SetBusy4
SetBusy5
SetBusy6
ServUp1
ServUp2
ServUp3
ServUp4
ServUp5
ServUp6
"Record"
. LL[20]>=LL[2]
42
. LL[20]<LL[2]
41
Chk3
LL[2]
Sort3
. LL[11]>0||LL[12]>0||LL[13]>0||LL[14]>0||LL[15]>0||LL[16]>0
70
"Done1"
75
137/30.5
0.0
100
BootServ
LL[10] = 1
Inc2
. LL[10]>6
LL[10] = LL[10]+1
Inc1
LL[21] = LL[10]
Inc3
Review
Release
.001
Check
Done1
80
Hold
INF
Done3



. NNACT(76)>0
77
"Done1"
.001
76
Record
"serve.txt"
TNOW
LL[10]
LL[11]
LL[12]
LL[13]
LL[14]
LL[15]
LL[16]
LL[20]
LL[21]
LL[22]
LL[23]
LL[24]
LL[25]
LL[26]
ChkStart
XX[51] . LL[10+LTRIB[1]]>0&&LL[21]==LTRIB[1]
21
Needed
LL[6] = LL[6]+1
PChg1
LTRIB[1]
ServUp



LL[4] = LL[4]-1
1
DETOUR3
LL[5]
POSITIVE
.5
.1
1
1stServ
TNOW
"Time 1st Service"
1
61
1stServ2
XX[55]
"Time Last Service"
1
62
LstServ2
LL[6]
"Plane Changes"
1
63
PChgs
INF
400
1
4
XX[52]/33
"Avg Improve"
1
64
Improved
(XX[60]+3*LL[9])/(LL[9]+LL[5])
"Mean Repair Time"
1
71
MeanRep
LL[5]
"Servicing Missions"
1
65
ServMsns
LL[9]
"Missed Servicings"
1
66
Missed
LL[32]
"Array Serves"
1
69
2Servs
LL[33]
"Clock Serves"
1
70
3Servs
LL[4]
POSITIVE
23.5
.1
1
SysUp
TNOW
"24+ Satellites"
1
13
SysUp2
LL[4]
NEGATIVE
23.5
.1
1
SysDown
TNOW
"24- Satellites"
1
14
SysDown2
4
INF
Done2

Appendix X: Explanation of Value Model Spreadsheet

We used an Excel spreadsheet to represent the value model and alternatives' performance parameters. The spreadsheet consists of several worksheets. The center of these is the "Overall Value Function" worksheet. It contains all the information necessary to determine total value scores for each alternative and to rank the alternatives. The "Single Dimensional Value Functions" portion contains references to the value function parameters of each measure. These measures came directly from the value model. We did not include the cost measures in the value model spreadsheet, because we compared value to cost separately. Seeing cost in terms of dollars instead of converting it to value is more meaningful to this particular sponsor. The value function parameters come from separate worksheets that correspond to each individual measure. For example, the "Sat Life" measure has a corresponding "Sat Life" worksheet.

We used two value function models in the spreadsheet. The first model was the boolean model. This model was appropriate when there were only two possible levels for a measure scale. This was the case in the "3 or 6" measure, which captured the impact of alternatives that required transitioning the constellation from six planes to three. An alternative could score a three for requiring transition and, thus, earn a value of 0. The only other possibility was for the alternative to score a six for not requiring a change and earn a value of 1. The function we used to reflect this value function was a piecewise linear function macro called ValuePL. In the case of the boolean value function, the piecewise linear function was a 1-piece linear function. The macro handled this situation just as well as it handled piecewise linear functions with multiple pieces. The ValuePL

function uses basic slope and line equations to calculate the value for an alternative's score.

The second model was the piecewise linear function. We used this for measures where it was necessary to elicit several discrete levels and corresponding values. The method of elicitation was first to ask for minimum and maximum practical levels. A practical minimum is a level below which an alternative does not lose or gain any additional value. The same applies to a practical maximum. The next step was to assess one or more midvalues. A midvalue is the level (we will call it an interlevel) at which the change in value between the lower level of the interval and the interlevel is the same change in value between the interlevel and the upper level of the interval. For example, the Capability measure could have a minimum meaningful level of 0% and a maximum of 90%. Those scores would correspond, on a 0 to 100 value scale, to values of 0 and 100 respectively. Using this full range of the scores as our interval, the decision-maker might assess 30% as the midvalue. Thus, 30% would receive a value of 50. If the decision-maker feels that higher resolution in the value function is necessary, the analyst can assess additional midvalues. Continuing with the example above, the decision-maker could now assess a midvalue between 30% and 90%.

After assessing the value functions, the next step was to weight the measures. The method we used for weighting was relative comparisons of each measure to one particular measure. We asked the decision-maker to specify the relative importance of each measure to the first. Thus, each measure would be a certain multiple of importance relative to the first measure. For example, the Capability measure could be 3 times as important as the Capacity measure and the 3 or 6 measure could be half as important as

the capacity measure. We then determined the individual weights using algebra and the knowledge that all the weights must add to one. Each measure has its own weight sensitivity analysis section lower in the Overall Value Function worksheet. We generated the sensitivity analysis numbers using Excel's Data Table feature. The reason for two lines of the weight data was to accommodate the Data Table feature. The sensitivity analysis determined the sensitivity of the final alternative ranking to the weighting of the measures. To do this, for each measure we needed to write the remaining measure weights in terms of the measure of interest. We have separate lines for each measure to avoid having to modify the calculations each time that we wanted to reaccomplish the sensitivity analysis. To write the weights in terms of a weight of interest, we used the method of maintaining the ratio relationships between the weights. The calculation was as follows:

$$w_x = (1 - w_i) * \left(\frac{w_x^o}{1 - w_i^o} \right) \qquad \text{Equation 34}$$

where w_i is the independent weight we are varying, w_x is the dependent weight we wish to adjust, and the weights with the superscript "o" are the original weights. These original weights preserve the ratio we are trying to maintain. Thus, every line of weights corresponding to a measure below the "Base Weights" line uses the above formula except for the cells corresponding to the measure of interest. Those weights use the "Base Weights" entry in their column and serve as the w_i value. We then created the first row and first column of a sensitivity analysis table for each measure. The row contains weights that would substitute for the w_i value in the above calculation. For this purpose, we used numbers from 0 to 1 in 0.05 increments. The first column began one cell below and to the left of the 0 in the first row. In the column, for each alternative, we placed the

calculation that summed the Weighted Single Dimensional Value Function data to yield the overall value. This summation was what we wanted Excel to recalculate for each value of that measure's weight from the first row we created above. Highlighting this row and column, we selected "Table..." from the "Data" menu. We entered the corresponding w_i for that measure from the diagonal of the Weights data above, and clicked "OK." Excel then filled in the table giving the new overall value function scores for the sample weight according to the column of the table. We plotted the results of each of these tables to better understand the data and to identify the range of weights for each measure that the highest ranked alternative would keep its ranking.

The "Probabilities and Scores for each Alternative" section of the worksheet contains the raw performance data for each alternative. Each score column is paired with a probability column. These probabilities correspond to the likelihood that each score will happen. The probability values are all one, because data did not exist on the distributions for possible outcomes in our measures. The interactions of the evolving technologies our alternatives utilized were unknown. Thus our evaluation was limited to an assumption of certainty in the outcomes. However, we designed the spreadsheet such that future users could easily add uncertainty considerations.

The "Weighted Single Dimensional Value Functions" section is the weight of the measure times the sumproduct of the scores and probability for the alternative in that measure. These results sum by alternative to get the final overall value. These values then make it possible to rank the alternatives. The end of the worksheet contains the sensitivity analysis data tables on the weights.

Appendix Y: Aerospace Study Progress

Table III. Summary of Configuration Design Features

Satellite Config.	Modifications
Basic Bus with Upgrade Compartment	*Additional Elements* Upgrade compartment, including support frame, structural panels, thrust cylinder, intercostals Removeable panels Connectors for removeable panels Wiring and ports for upgrade components *Modifications to Baseline Structures* Interface hardware for upgrade compartment *RS Specific Features* Docking rings Grapple fixtures
Basic Bus with Upgrade Boom	*Additional Elements* Deployable boom Connectors for upgrade slots Wiring and ports for upgrade components *Modifications to Baseline Structures* Structural reinforcement for boom attachments Interface hardware for boom *RS Specific Features* Docking rings
Reconfigured Equipment Panels Removeable Subpanels	*Additional Elements* Frame structure for removeable panels Removeable panels Connectors for removeable panels Wiring and ports for upgrade components *Modifications to Baseline Structures* Stretched thrust cylinder, intercostals, and side panels *RS Specific Features* Docking rings Grapple fixtures
Reconfigured Equipment Panels Drawers	*Additional Elements* Frame structure and slide rails for drawers Drawers Connectors for ORUs Wiring and ports for ORUs *Modifications to Baseline Structures* Stretched thrust cylinder, intercostals, and side panels *RS Specific Features* Docking rings

	Grapple fixtures
Reconfigured Equipment Panels Access doors	*Additional Elements* Frame structure for doors Latches and Hinges Flexible cables at hinges Connectors for ORUs Wiring and ports for ORUs *Modifications to Baseline Structures* Interface hardware for door hinges *RS Specific Features* Docking rings Grapple fixtures

Mass Impact From Structures

The mass impact for the various concepts were estimated using a combination of sizing algorithms and extrapolation from existing hardware. Where necessary, finite element analysis was also performed to assess feasibility. Detailed mass breakdowns are provided in the Appendix. The total structures weight impact is summarized in Table IV.

Table IV. Structures Mass Impact for Serviceability

Satellite Config.	**High Capability RS**	**Medium Capability RS**	**Low Capability RS**
	Mass Impact, lb	**Mass Impact, lb**	**Mass Impact, lb**
Basic Bus with 20 inch Upgrade Compartment	99	127	103
Basic Bus with 40 inch Upgrade Compartment	152	180	157
Basic Bus with Upgrade Boom	37	37	37
Reconfigured Equipment Panels Removeable Subpanels	184	212	200
Reconfigured Equipment Panels Drawers	287	314	N/A
Reconfigured Equipment Panels Access doors	200	227	N/A

System Impact

Based on the delta-mass from satellite reconfigurations, the total system impact was estimated from a GPS sizing tool developed by the Vehicle Concepts Department. Results are summarized in Table V-VII. **Note that the baseline wet mass is 2813 lb.**

Next, the system weight was provided to the Cost and Requirements Department to estimate the total system impact. Results are summarized in Table V-VII.

Table V. System Impacts of Reconfiguration Options for High Capability RS

Reconfiguration Option	System Mass	System Cost	% Serviceable	# Upgrade Slots
Basic Bus with 20 inch Upgrade Compartment	3150		0% Upgrades Only	4 medium sized electronic boxes
Basic Bus with 40 inch Upgrade Compartment	3488		0% Upgrades Only	4 large or 8 medium sized electronics boxes
Basic Bus with Upgrade Boom	3065		0% Upgrades Only	3 medium sized electronic boxes
Reconfigured Equipment Panels Removeable Subpanels	3225		40% serviceable	3 upgrade slots, but only for thermally insenstive components

Table VI. System Impacts of Reconfiguration Options for Medium Capability RS

Reconfiguration Option	System Mass	System Cost	% Serviceable	# Upgrade Slots
Basic Bus with 20 inch Upgrade Compartment	3189		0% Upgrades Only	4 medium sized electronic boxes
Basic Bus with 40 inch Upgrade Compartment	3519		0% Upgrades Only	4 large or 8 medium sized electronics boxes
Basic Bus with Upgrade Boom	3077		0% Upgrades Only	3 medium sized electronic boxes
Reconfigured Equipment Panels	3261		40% serviceable	3 upgrade slots, but only for thermally

Removeable Subpanels				insenstive components

Table VII. System Impacts of Reconfiguration Options for Low Capability RS

Reconfiguration Option	**System Mass**	**System Cost**	**% Serviceable**	**# Upgrade Slots**
Basic Bus with 20 inch Upgrade Compartment	3160		0% Upgrades Only	4 medium sized electronic boxes
Basic Bus with 40 inch Upgrade Compartment	3491		0% Upgrades Only	4 large or 8 medium sized electronics boxes
Basic Bus with Upgrade Boom	3066		0% Upgrades Only	3 medium sized electronic boxes
Reconfigured Equipment Panels Removeable Subpanels	3246		40% serviceable	3 upgrade slots, but only for thermally insenstive components

Appendix Z: Total Cost Tables

Spreadsheet 23: TOTAL COST (page 1)

Architectures	A (or C)											
Alternatives	1	2	3	4	5	6	7	8	9	10	11	12
Number of GPS planes =	6	3	3	3	6	3	3	3	6	6	6	3
ORU mass (kg) =	300	300	150	300	150	150	150	50	50	50	50	300
Robotic Servicer	High	High	High	Medium	Medium	Medium	Low	Low	Low	Low	Low	High
Number of service times =	1	1	1	1	1	1	1	1	1	1	1	4
Solar thermal transfer time =	25 days			Chemical transfer time= 1 day				Precession Period in LEO for ORU canisters =				20 days
RS Design life =	120 days	120 days	120 days	120 days	120 days	120 days	120 days	120 days	120 days	120 days	120 days	15 years
Cycle Time Servicing	60 days	120	120	120	60	120	120	120	60	60	1095	132
Total Cycle Time	105	165	165	165	105	165	145	165	81	105	1095	177
RS Mass (w/o propul.) =	378	378	378	203	203	203	135	135	135	135	135	494
R/S Mass Total =	530	620	540	478	314	360	301	221	183	383	383	937
LV / Upper St. (R.S.)	M / solid	M /sol th	M / solid	Del /sol th	M /sol th	Del /sol th	Del /sol th.	Del / liquid	M / solid	M / sol th	Piggyback	M+/ sol th.
LV / Upper St. (ORU)	M+ / sol th	M+ / sol th	M+ / sol th	M+ / sol th	M /sol th	M+ /sol th	M+ /sol th	Del / sol th	M / solid	N/A	N/A	M+ / sol th
notes	Both 2x	ORU 2x		ORU 2x	ORU 2x					combined	combined	ORU 8x
											Up.S. & dis like M (3 OTC)	
COST (total [1999 Dollars in Millions])												
Reoccuring												
R.S.												
Launch	146	73	73	49	73	49	49	49	73	73	N/A	95
Up. St./Disp	38	14	19	9	11	9	10	10	13.5	17.5	60	18.5
R.S.	104	52	52	41	81	41	26	26	51	51	51	81
(note: R.S. includes system integration & program management)												
ORU												
Launch	190	190	95	190	146	95	95	49	73			760
Up. St./Disp/Can	44	44	22	44	16	22	22	11.5	21.5			144
1st mission total	522	373	261	333	327	216	202	146	232	142	111	421
Total												1,099
Average cost / mission												275
RDT&E	(the $2M below because R.S. & ORU have the same Up. St.)							(see note to the side)				
R.S.												
Up. St./Disp	7	2	7	14.5	2	14.5	15	10	2	0	7.4	2
R.S.	77.5	77.5	77.5	56.5	56.5	56.5	44.5	44.5	44.5	44.5	44.5	111.4
ORU Up. S./Disp/Can	34	34	34	34	23.5	34	34	14.5	12	15.5	0	34
Total	118.5	113.5	118.5	105	82	105	93.5	69	58.5	60	51.9	147.4
1st mission Total	641	487	380	438	409	321	296	215	291	202	163	568
4 mission Total	2,207	1,606	1,163	1,437	1,390	969	902	651	987	626	496	1,246
average mission	552	401	291	359	348	242	225	163	247	157	124	311

TOTAL COST (page 2)

B (or C)									D			E	
13	14	15	16	17	18	19	20		21	22		23	24
6	3	3	6	3	3	3	6		3	6		6	6
150	150	50	300	150	50	150	50		150(u)20(R)	150(u)20(R)		150(u)20(R)	50(u)20(R)
High	High	High	Medium	Medium	Medium	Low	Low		High	High		med	low
4	4	4	4	4	4	4	4		<-------------- 4 upgrades, repair is continous (model by 7 servicings / S.				
15 years	15 years	15 yrs	15 years	15 years	15 years	15 years	15 years		15 years	15 years		15 years	15 years
72	132	132	72	132	132	132	72		120	60		675	675
117	177	177	117	177	177	177	93		165	105		695	676
494	494	494	263	263	263	166	166		494	494		263	143
639	818	738	387	491	412	292	183		1021	715		623	506
											fuel =	11,470	8745
M /sol th	M / sol th.	M+ / solid	M / sol th	M / sol th	M / sol th	Delt / liq	M / liq		M+ / Sol th	M / sol th		M+ (1 time)	M+
M /sol th	M+ / sol th	Del / sol th	M+/sol th	M+/sol th	Del / sol th	M+/sol th	M / liq	upg. =	M+ / Sol th	M / sol th	fuel =	M+ (13 times)	M (13 times)
ORU 8x	ORU 4x	ORU 4x	ORU 8x	ORU 4x	ORU 4x	ORU 4x	ORU 4x	depot=	M+ / Sol th	M / sol th	upg. =	M /sol th(8x)	M / liq (4x)
R.S. 2x			R.S. 2x						upg 4x	up 8x /rest 2x	depot=	M (one time)	M (one time)
												<--------- (R.S. launch includes f	
146	73	95	146	73	73	49	73		95	146		1330	1044
26	13	10	26	13	13	10	13.5		18.5	22		none	none
162	81	81	92	46	46	30	60		81	162		20	15
											tank =	6	5
584	380	196	760	380	196	380	292		475	730		657	365
104	80	42	144	80	42	80	58		100	155		104	62
506	282	246	490	247	192	204	234		308	414		461	348
1,022	627	424	1,168	592	370	500	424		770	1,215		2,117	1,491
256	157	106	292	148	93	125	106		192	304		529	373
											tank =	10.9	9.9
2	2	18	2	2	2	10	2		2	2		none	none
111.4	111.4	111.4	64.6	64.6	64.6	52.5	52.5		111.4	111.4		86.6	74.5
23.5	34	14.5	34	34	14.5	34	12		34	23.5		23.5	11.8
136.9	147.4	143.9	100.6	100.6	81.1	96.5	66.5		147.4	136.9		121	96.2
643	429	389	591	348	273	301	301		455	551		582	444
1,159	774	568	1,269	693	451	597	490		917	1,352		2,238	1,587
290	194	142	317	173	113	149	123		229	338		560	397

F		G				H
	25	26	27	28	29	30
	6	*6*	*6*	*6*	*6*	*6*
50(u)20(R)		*300*	*150*	*150*	*50*	*150*
low		*med*	*med*	*low*	*low*	*med*
;/V)--------------->		*1*	*1*	*1*	*1*	*1*
15 years		2 years	2 years	2 years	2 years	2 years
	675	1116	814	781	643	361
	676	1141	839	806	668	386
	143	217	217	143	143	220
	300	3000	1616	1519	864	3900
	8675					
M+		K-1 / ion	K-1 / ion	K-1 / ion	Taurus / ion	M + /sol th
M (13 times)		M+ /sol th	M /sol th	M /sol th	M /sol th	M /sol th
M / liq (4x)		ORU 2x	ORU 2x	ORU 2x	ORU 2x	ORU 2x
M (one time)						
fuel) --------->						
	1044	*33*	*33*	*33*	*25*	*93*
none		*none*	*none*	*none*	*none*	*none*
	10	*21.6*	*21.6*	*16.4*	*16.4*	*18*
	4					
	365	*190*	*146*	*146*	*146*	*146*
	62	*44*	*32*	*32*	*32*	*32*
	259	289	233	227	219	289
	1,485					
	371					
	7.5					
none		*none*	*none*	*none*	*none*	*none*
	52.5	*83.8*	*83.8*	*70.5*	*70.5*	*77.8*
	12	*34*	*23.5*	*23.5*	*23.5*	*23.5*
	72	117.8	107.3	94	94	101.3
	331	406	340	321	313	390
	1,557	1,272	1,038	1,004	972	1,257
	389	318	259	251	243	314

Appendix AA: MathCAD Means Test Worksheets

$$V := \begin{bmatrix} 8.583809524 \\ 7.164761905 \\ 6.326666667 \\ 6.703809524 \\ 7.347619048 \\ 5.875238095 \\ 5.421428571 \\ 4.896666667 \\ 6.345238095 \\ 6.342380952 \\ 6.011489177 \\ 7.576380952 \\ 8.196380952 \\ 6.747809524 \\ 6.183047619 \\ 8.616952381 \\ 6.359809524 \\ 5.82552381 \\ 5.891904762 \\ 6.806190476 \\ 7.425058201 \\ 8.87362963 \\ 7.911139971 \\ 6.919893218 \\ 6.965988456 \\ 7.977264069 \\ 7.312415584 \\ 6.877619048 \\ 6.371991342 \\ 7.538562771 \end{bmatrix}$$

The "V" vector contains the overall value of Alternatives 1 thorugh 30

Determine if statistical difference in value exists between 6-servicer and 1-servicer alternatives

6-servicer alternatives	1-servicer alternatives
$Serv6 := submatrix(V,1,22,1,1)$	$Serv1 := submatrix(V,23,30,1,1)$
$n6 := rows(Serv6)$	$n1 := rows(Serv1)$
$S6 := stdev(Serv6)$	$S1 := stdev(Serv1)$
$Y6 := mean(Serv6) \quad Y6 = 6.796$	$Y1 := mean(Serv1) \quad Y1 = 7.234$

$$S_p := \sqrt{\frac{(n6-1)\cdot S6^2 + (n1-1)\,S1^2}{n6+n1-2}} \qquad S_p = 0.944$$

95% confidence interval with t-statistic and n6+n1-2 DOF

$$DOF := n6 + n1 - 2 \qquad DOF = 28 \qquad \alpha := 0.05$$

$$t := qt\left(1 - \frac{\alpha}{2}, DOF\right) \qquad t = 2.048$$

$$LL := (Y1 - Y6) - t\cdot S_p\cdot\sqrt{\frac{1}{n6}+\frac{1}{n1}} \qquad LL = -0.36$$

$$UL := (Y1 - Y6) + t\cdot S_p\cdot\sqrt{\frac{1}{n6}+\frac{1}{n1}} \qquad UL = 1.236$$

The limits do encompass zero, therefore there is not a statistically significant difference between the means. Thus, the 1-servicer options do not perform differently from the 6-servicer options. Note: I used "6-servicer" to imply all alternatives with one servicer per plane.

Determine if statistical difference in value exists between 6-plane and 3-plane alternatives

The "submatrix" and "stack" statements assemble the appropriate data into vectors.

6-plane alternatives	3-plane alternatives
Plane6 := submatrix(V, 1, 1, 1, 1)	Plane3 := submatrix(V, 2, 4, 1, 1)
TEMP6 := submatrix(V, 5, 5, 1, 1)	TEMP3 := submatrix(V, 6, 8, 1, 1)
Plane6 := stack(Plane6, TEMP6)	Plane3 := stack(Plane3, TEMP3)
TEMP6 := submatrix(V, 9, 11, 1, 1)	TEMP3 := submatrix(V, 12, 12, 1, 1)
Plane6 := stack(Plane6, TEMP6)	Plane3 := stack(Plane3, TEMP3)
TEMP6 := submatrix(V, 13, 13, 1, 1)	TEMP3 := submatrix(V, 14, 15, 1, 1)
Plane6 := stack(Plane6, TEMP6)	Plane3 := stack(Plane3, TEMP3)
TEMP6 := submatrix(V, 16, 16, 1, 1)	TEMP3 := submatrix(V, 17, 19, 1, 1)
Plane6 := stack(Plane6, TEMP6)	Plane3 := stack(Plane3, TEMP3)
TEMP6 := submatrix(V, 20, 20, 1, 1)	TEMP3 := submatrix(V, 21, 21, 1, 1)
Plane6 := stack(Plane6, TEMP6)	Plane3 := stack(Plane3, TEMP3)
TEMP6 := submatrix(V, 22, 30, 1, 1)	
Plane6 := stack(Plane6, TEMP6)	

n6 := rows(Plane6) S6 := stdev(Plane6) Y6 := mean(Plane6) Y6 = 7.353

n3 := rows(Plane3) S3 := stdev(Plane3) Y3 := mean(Plane3) Y3 = 6.338

$$S_p := \sqrt{\frac{(n6-1)\cdot S6^2 + (n3-1)\, S3^2}{n6+n3-2}}$$

$S_p = 0.814$

95% confidence interval with t-statistic and n6+n3-2 DOF

$DOF := n6 + n3 - 2$ $DOF = 28$ $\alpha := 0.05$

$t := qt\left(1 - \frac{\alpha}{2}, DOF\right)$ $t = 2.048$

$$LL := (Y6 - Y3) - t\cdot S_p\cdot\sqrt{\frac{1}{n6}+\frac{1}{n3}} \qquad LL = 0.4$$

$$UL := (Y6 - Y3) + t\cdot S_p\cdot\sqrt{\frac{1}{n6}+\frac{1}{n3}} \qquad UL = 1.629$$

The limits do not encompass zero, therefore there is a statistically significant difference between the means. Thus, the 6-plane options outperform the 3-plane options

Determine if statistical difference in value exists between alternatives with different ORU capacities

Low capacity (50 kg)

Low := submatrix(V, 8, 11, 1, 1)

TEMPL := submatrix(V, 15, 15, 1, 1)

Low := stack(Low, TEMPL)

TEMPL := submatrix(V, 18, 18, 1, 1)

Low := stack(Low, TEMPL)

TEMPL := submatrix(V, 20, 20, 1, 1)

Low := stack(Low, TEMPL)

TEMPL := submatrix(V, 24, 25, 1, 1)

Low := stack(Low, TEMPL)

TEMPL := submatrix(V, 29, 29, 1, 1)

Low := stack(Low, TEMPL)

Med capacity (150 kg)

Med := submatrix(V, 3, 3, 1, 1)

TEMPM := submatrix(V, 5, 7, 1, 1)

Med := stack(Med, TEMPM)

TEMPM := submatrix(V, 13, 14, 1, 1)

Med := stack(Med, TEMPM)

TEMPM := submatrix(V, 17, 17, 1, 1)

Med := stack(Med, TEMPM)

TEMPM := submatrix(V, 19, 19, 1, 1)

Med := stack(Med, TEMPM)

TEMPM := submatrix(V, 21, 23, 1, 1)

Med := stack(Med, TEMPM)

TEMPM := submatrix(V, 27, 28, 1, 1)

Med := stack(Med, TEMPM)

TEMPM := submatrix(V, 30, 30, 1, 1)

Med := stack(Med, TEMPM)

High capacity (300 kg)

High := submatrix(V, 1, 2, 1, 1)

TEMPH := submatrix(V, 4, 4, 1, 1)

High := stack(High, TEMPH)

TEMPH := submatrix(V, 12, 12, 1, 1)

High := stack(High, TEMPH)

TEMPH := submatrix(V, 16, 16, 1, 1)

High := stack(High, TEMPH)

TEMPH := submatrix(V, 26, 26, 1, 1)

High := stack(High, TEMPH)

$nLow := rows(Low)$ $\quad$ $SLow := stdev(Low)$ $\quad$ $YLow := mean(Low)$ $\quad$ $YLow = 6.267$

$nMed := rows(Med)$ $\quad$ $SMed := stdev(Med)$ $\quad$ $YMed := mean(Med)$ $\quad$ $YMed = 7.008$

$nHigh := rows(High)$ $\quad$ $SHigh := stdev(High)$ $\quad$ $YHigh := mean(High)$ $\quad$ $YHigh = 7.77$

Test for difference between 50 kg and 150 kg capacity alternatives

$$S_p := \sqrt{\frac{(nLow - 1)\cdot SLow^2 + (nMed - 1)\ SMed^2}{nMed + nLow - 2}} \qquad S_p = 0.813$$

95% confidence interval with t-statistic and nMed+nLow-2 DOF

$DOF := nLow + nMed - 2$ $\quad$ $DOF = 22$ $\quad$ $\alpha := 0.05$ $\quad$ $t := qt\left(1 - \frac{\alpha}{2}, DOF\right)$ $\quad$ $t = 2.074$

$$LL := (YMed - YLow) - t\cdot S_p\cdot\sqrt{\frac{1}{nMed} + \frac{1}{nLow}} \qquad LL = 0.043$$

$$UL := (YMed - YLow) + t\cdot S_p\cdot\sqrt{\frac{1}{nMed} + \frac{1}{nLow}} \qquad UL = 1.439$$

The limits do not encompass zero, therefore there is a statistically significant difference between the means. Thus, the 150 kg options outperform the 50 kg options.

Test for difference between 300 kg and 150 kg capacity alternatives

$$S_p := \sqrt{\frac{(nHigh - 1)\cdot SHigh^2 + (nMed - 1)\ SMed^2}{nMed + nHigh - 2}} \qquad S_p = 0.881$$

95% confidence interval with t-statistic and nMed+nHigh-2 DOF

$DOF := nHigh + nMed - 2$ $\quad$ $DOF = 18$ $\quad$ $\alpha := 0.05$ $\quad$ $t := qt\left(1 - \frac{\alpha}{2}, DOF\right)$ $\quad$ $t = 2.101$

$$LL := (YHigh - YMed) - t\cdot S_p\cdot\sqrt{\frac{1}{nMed} + \frac{1}{nHigh}} \qquad LL = -0.14$$

$$UL := (YHigh - YMed) + t\cdot S_p\cdot\sqrt{\frac{1}{nMed} + \frac{1}{nHigh}} \qquad UL = 1.666$$

The limits do encompass zero, therefore there is not a statistically significant difference between the means. Thus, the 300 kg capacity options cannot be said to differ from the 150 kg options.

Test for difference between 300 kg and 50 kg capacity alternatives

$$S_p := \sqrt{\frac{(nHigh - 1)\cdot SHigh^2 + (nLow - 1)\, SLow^2}{nLow + nHigh - 2}} \qquad S_p = 0.627$$

95% confidence interval with t-statistic and nLow+nHigh-2 DOF

$$DOF := nHigh + nLow - 2 \qquad DOF = 14 \qquad \alpha := 0.05 \qquad t := qt\left(1 - \frac{\alpha}{2}, DOF\right) \qquad t = 2.145$$

$$LL := (YHigh - YLow) - t\cdot S_p\cdot\sqrt{\frac{1}{nLow} + \frac{1}{nHigh}} \qquad LL = 0.809$$

$$UL := (YHigh - YLow) + t\cdot S_p\cdot\sqrt{\frac{1}{nLow} + \frac{1}{nHigh}} \qquad UL = 2.198$$

The limits do not encompass zero, therefore there is a statistically significant difference between the means. Thus, the 300 kg capacity options outperform the 50 kg options.

$$MH := stack(Med, High)$$

$$nMH := rows(MH) \qquad SMH := stdev(MH) \qquad YMH := mean(MH) \qquad YMH = 7.236$$

Test for difference between 50 kg capacity alternatives and the combined 300 kg and 150 kg capacity alternatives

$$S_p := \sqrt{\frac{(nLow - 1)\cdot SLow^2 + (nMH - 1)\, SMH^2}{nMH + nLow - 2}} \qquad S_p = 0.844$$

95% confidence interval with t-statistic and nMH+nLow-2 DOF

$$DOF := nLow + nMH - 2 \qquad DOF = 28 \qquad \alpha := 0.05 \qquad t := qt\left(1 - \frac{\alpha}{2}, DOF\right)$$

$$LL := (YMH - YLow) - t\cdot S_p\cdot\sqrt{\frac{1}{nMH} + \frac{1}{nLow}} \qquad LL = 0.3$$

$$UL := (YMH - YLow) + t\cdot S_p\cdot\sqrt{\frac{1}{nMH} + \frac{1}{nLow}} \qquad UL = 1.639$$

The limits do not encompass zero, therefore there is a statistically significant difference between the means. Thus, the 50 kg options underperform the 150 kg and 300 kg options.

Determine if statistical difference in value exists between alternatives with different servicer capabilities

Low capability

Low := submatrix(V, 7, 11, 1, 1)

TEMPL := submatrix(V, 19, 20, 1, 1)

Low := stack(Low, TEMPL)

TEMPL := submatrix(V, 24, 25, 1, 1)

Low := stack(Low, TEMPL)

TEMPL := submatrix(V, 28, 29, 1, 1)

Low := stack(Low, TEMPL)

Med capability

Med := submatrix(V, 4, 6, 1, 1)

TEMPM := submatrix(V, 16, 18, 1, 1)

Med := stack(Med, TEMPM)

TEMPM := submatrix(V, 23, 23, 1, 1)

Med := stack(Med, TEMPM)

TEMPM := submatrix(V, 26, 27, 1, 1)

Med := stack(Med, TEMPM)

TEMPM := submatrix(V, 30, 30, 1, 1)

Med := stack(Med, TEMPM)

High capability

High := submatrix(V, 1, 3, 1, 1)

TEMPH := submatrix(V, 12, 15, 1, 1)

High := stack(High, TEMPH)

TEMPH := submatrix(V, 21, 22, 1, 1)

High := stack(High, TEMPH)

$nLow := rows(Low)$ $SLow := stdev(Low)$ $YLow := mean(Low)$ $YLow = 6.259$

$nMed := rows(Med)$ $SMed := stdev(Med)$ $YMed := mean(Med)$ $YMed = 7.147$

$nHigh := rows(High)$ $SHigh := stdev(High)$ $YHigh := mean(High)$ $YHigh = 7.453$

Test for difference between low and medium capability alternatives

$$S_p := \sqrt{\frac{(nLow - 1)\cdot SLow^2 + (nMed - 1)\, SMed^2}{nMed + nLow - 2}} \qquad S_p = 0.763$$

95% confidence interval with t-statistic and nMed+nLow-2 DOF

$$DOF := nLow + nMed - 2 \quad DOF = 19 \quad \alpha := 0.05 \quad t := qt\left(1 - \frac{\alpha}{2}, DOF\right) \quad t = 2.093$$

$$LL := (YMed - YLow) - t\cdot S_p\cdot\sqrt{\frac{1}{nMed} + \frac{1}{nLow}} \quad LL = 0.19$$

$$UL := (YMed - YLow) + t\cdot S_p\cdot\sqrt{\frac{1}{nMed} + \frac{1}{nLow}} \quad UL = 1.585$$

The limits do not encompass zero, therefore there is a statistically significant difference between the means. Thus, the medium capability options outperform the low capability options.

Test for difference between medium and high capability alternatives

$$S_p := \sqrt{\frac{(nHigh - 1)\cdot SHigh^2 + (nMed - 1)\, SMed^2}{nMed + nHigh - 2}} \qquad S_p = 0.893$$

95% confidence interval with t-statistic and nMed+nHigh-2 DOF

$$DOF := nHigh + nMed - 2 \quad DOF = 17 \quad \alpha := 0.05 \quad t := qt\left(1 - \frac{\alpha}{2}, DOF\right) \quad t = 2.11$$

$$LL := (YHigh - YMed) - t\cdot S_p\cdot\sqrt{\frac{1}{nMed} + \frac{1}{nHigh}} \quad LL = -0.56$$

$$UL := (YHigh - YMed) + t\cdot S_p\cdot\sqrt{\frac{1}{nMed} + \frac{1}{nHigh}} \quad UL = 1.172$$

The limits do encompass zero, therefore there is not a statistically significant difference between the means. Thus, the high capability options cannot be said to differ from the medium capability options.

Test for difference between low and high capability alternatives

$$S_p := \sqrt{\frac{(nHigh - 1)\cdot SHigh^2 + (nLow - 1)\, SLow^2}{nLow + nHigh - 2}} \qquad S_p = 0.764$$

95% confidence interval with t-statistic and nLow+nHigh-2 DOF

$$DOF := nHigh + nLow - 2 \qquad DOF = 18 \qquad \alpha := 0.05 \qquad t := qt\left(1 - \frac{\alpha}{2}, DOF\right) \qquad t = 2.101$$

$$LL := (YHigh - YLow) - t\cdot S_p \cdot \sqrt{\frac{1}{nLow} + \frac{1}{nHigh}} \qquad LL = 0.472$$

$$UL := (YHigh - YLow) + t\cdot S_p \cdot \sqrt{\frac{1}{nLow} + \frac{1}{nHigh}} \qquad UL = 1.916$$

The limits do not encompass zero, therefore there is a statistically significant difference between the means. Thus, the high capability options outperform the low capability options.

$$MH := stack(Med, High)$$

$$nMH := rows(MH) \qquad SMH := stdev(MH) \qquad YMH := mean(MH) \qquad YMH = 7.292$$

Test for difference between low capability alternatives and the combined high and medium capability alternatives

$$S_p := \sqrt{\frac{(nLow - 1)\cdot SLow^2 + (nMH - 1)\, SMH^2}{nMH + nLow - 2}} \qquad S_p = 0.819$$

95% confidence interval with t-statistic and n6+n1-2 DOF

$$DOF := nLow + nMH - 2 \qquad DOF = 28 \qquad \alpha := 0.05 \qquad t := qt\left(1 - \frac{\alpha}{2}, DOF\right)$$

$$LL := (YMH - YLow) - t\cdot S_p \cdot \sqrt{\frac{1}{nMH} + \frac{1}{nLow}} \qquad LL = 0.397$$

$$UL := (YMH - YLow) + t\cdot S_p \cdot \sqrt{\frac{1}{nMH} + \frac{1}{nLow}} \qquad UL = 1.668$$

The limits do not encompass zero, therefore there is a statistically significant difference between the means. Thus, the low capability options underperform the medium and high capability options.

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13. ABSTRACT *(Maximum 200 words)*
Satellites are the only major Air Force systems with no maintenance, routine repair, or upgrade capability. The result is expensive satellites and a heavy reliance on access to space. At the same time, satellite design is maturing which is reducing the cost to produce satellites with long design lives. This works against the ability to keep the technology on satellites current without frequent replacement of those satellites. The Global Positioning System (GPS) Joint Program Office (JPO) realizes that it must change its mode of operations to quickly meet new requirements while minimizing cost.
This thesis looks at the possibility of using robotic servicing architectures to solve these problems. It accomplished this through a systems engineering and decision analysis approach. It analyzed different space systems alternatives for satellite repair and upgrade. This approach involved defining the problem framework and desired user benefits. It developed different system architectures and determined their performance. Finally, the authors used decision analysis to evaluate the alternative architectures in the context of the user's goals. The results indicated favorable benefit-to-cost relationships of on-orbit servicing architectures.

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